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EXPERIMENTAL INVESTIGATION OF THE CRACK GROWTH GAGE

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Structural Integrity Branch
Structures & Dynamics Division

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Final Report for Period January 1978 to November 1979

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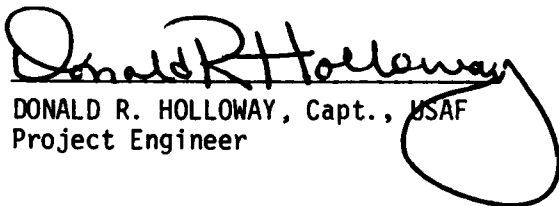
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This technical report has been reviewed and is approved for publication.


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gage. Inconsistencies in constant amplitude test results and bonding problems encountered on the F-4C/D full-scale fatigue test article prevented the MIL-STD-810C qualification tests from being started. Results of this investigation indicated that further research and development of the crack growth gage concept is required before the gage can be recommended as a fleetwide tracking device.

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FOREWORD

This report describes an in-house effort conducted under Project 2401, "Structures and Dynamics," Task 240101, "Structural Integrity for Military Aerospace Vehicles," Work Unit 24010109, "Life Analysis and Design Methods for Aerospace Structure."

This work was performed for the Structural Integrity Branch, Structures and Dynamics Division, Air Force Flight Dynamics Laboratory (AFFDL/FBE). This organization is currently the Structural Integrity Branch, Structures and Dynamics Division, Flight Dynamics Laboratory, Air Force Wright Aeronautical Laboratories (AFWAL/FIBE), Wright-Patterson Air Force Base, Ohio. The research was conducted under the direction of Captain D.R. Holloway and Mr. T.D. Gray from January 1978 through November 1979.

The author wishes to recognize Messrs. Harold Stalnaker, Jack Smith, Richard Kleismit, and Larry Bates for their contributions in the accomplishment of the experimental phases of this study. In addition, the efforts of Mr. Jeff Wead for drafting the figures, Mr. Pete Dodaro for preparing the data plotting routines, and Mr. John Potter for his guidance throughout the program were much appreciated.

The completed report was submitted in December 1979.

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TABLE OF CONTENTS

SECTION		PAGE
I	INTRODUCTION	1
	1. Background	2
	2. Crack Growth Gage Technique for IAT	5
	3. Program Objective	11
II	TEST PROGRAM	13
	1. Introduction	13
	2. Test Materials	13
	2.1 Alloy Selection	13
	2.2 da/dN Coupons	13
	2.3 Crack Growth Gage Design	15
	2.4 Adhesive Selection	15
	3. Test Procedures	17
	3.1 Crack Growth Gage Bonding Technique	17
	3.2 Crack Monitoring	17
	3.3 Precracking of Crack Growth Gages	21
	3.4 Primary Specimen Testing	21
	3.5 da/dN Coupon Testing	21
	3.6 Constant Amplitude Testing	23
	3.7 Strain Gage Instrumentation	23
	4. Fatigue Test Article	23
	4.1 Fatigue Test Article Surface Preparation	30
	4.2 Bonding Process	30
III	PRESENTATION AND DISCUSSION OF EXPERIMENTAL RESULTS	33
	1. Constant Amplitude Test Results	33
	2. Fatigue Test Article Results	43
IV	CONCLUSIONS	53
V	RECOMMENDATIONS	55
	APPENDIX A da/dN Coupon Test Results	56
	APPENDIX B Strain Measurements	59
	APPENDIX C Constant Amplitude Test Results	64
	REFERENCES	90

LIST OF ILLUSTRATIONS

FIGURE		PAGE
1	Schematic Representation of Current F-4 Tracking Analysis Method Employing the Damage Index and Equivalent S-N Curve System	4
2	Schematic Representation of Damage Tracking Using the Crack Growth Gage Concept	7
3	Schematic View of Crack Growth Gage Attached to Flawed Structural Component	9
4	da/dN Coupon	14
5	Crack Growth Gage	16
6	Crack Growth Gage Bonding Technique	18
7	Example of Fax-Film Replication	20
8	Crack Growth Gage Precracking Configuration	22
9	Crack Growth Gage Test Configuration: Normal Length Gage-Bonded	24
10	Crack Growth Gage Test Configuration: Full Length Gage (With Precracking Tabs) - Bonded	25
11	Crack Growth Gage Test Configuration: Full Length Gage (With Precracking Tabs) - Bonded and Bolted	26
12	Selected Gage Sites	28
13	Relationship of Selected Gage Sites to Stress Contours on Lower Wing Skin	29
14	Vacuum Bag Technique of Bonding Crack Growth Gages	31
15	Constant Amplitude Tests - AFFDL Tests of AFFDL Gages Compared with McAir Tests of McAir Gages	34
16	Constant Amplitude Tests - AFFDL Tests of McAir Gages Compared with McAir Tests of McAir Gages	36
17	Constant Amplitude Tests - AFFDL Tests of AFFDL Full Length Gages (Precracking Tabs Attached) Compared with McAir Gages	38

LIST OF ILLUSTRATIONS (Cont'd)

FIGURE		PAGE
18	Constant Amplitude Tests - AFFDL Tests of AFFDL Gages Having Tighter Manufacturing Specifications Compared with McAir Tests of McAir Gages	39
19	Constant Amplitude Tests - McAir Tests of AFFDL Gages Compared with McAir Tests of McAir Gages	40
20	Selected Sites on the Lower Wing Skin of the F-4 Fatigue Article for Bonding Crack Growth Gages with EA9309.1 Adhesive	46
21	Comparison of Crack Length Measurements and Predictions for Site 1 (Site 5 on Left Wing)	48
22	Comparison of Crack Length Measurements and Predictions for Site 2 (Site 4 on Left Wing)	49
23	Comparison of Crack Length Measurements and Predictions for Site 3	50
24	Comparison of Crack Length Measurements and Predictions for Site 4	51
25	Comparison of Crack Length Measurements and Predictions for Site 7 (Site 8 on Left Wing)	52

SUMMARY

Before crack growth gages can be recommended to track aircraft service life, the performance and predictability of the gages must be verified. This requires engineering analysis and verification by test.

An on-going fatigue test of a full-scale F-4 C/D test article being conducted by the Air Force Flight Dynamics Laboratory provided a convenient test bed for evaluating the crack growth gage concept for use with actual aircraft. The purpose of the F-4 C/D full-scale fatigue test is to provide full-scale test verification of several life extension modifications, including those designed to extend the life to 8000 flight hours of F-4 ASIP baseline usage. During one modification implementation, while the test was in a hold status, crack growth gages were adhesively bonded to the test structure. Testing of the gages attached to the fatigue article was directed towards developing data that would verify that the gages would provide meaningful and predictable output for scheduling structural modifications, repairs, inspections, and retirement of individual F-4 airframes.

Besides the above major task, two additional tasks were required. These tasks consisted of (1) conducting a gage qualification test program in accordance with MIL-STD-810C (Environmental Test Methods) requirements, and (2) determining an appropriate method for collecting data from the gage. Inconsistencies in constant amplitude test results prevented the MIL-STD-810C qualification tests from being started. These inconsistencies, along with problems encountered with the bonding of the gages to the full-scale fatigue article, indicated that further research and development of the crack growth gage concept is required before the gage can be recommended as a fleet-wide tracking device.

SECTION I

INTRODUCTION

Maintaining the damage tolerance and durability of USAF aircraft structures is dependent on the capability of the appropriate Air Force Command to perform specific inspection, maintenance, and possibly modification or replacement tasks at specific intervals throughout the service life (i.e., at specified depot or base level maintenance times and special inspection periods). Experience has shown that the actual usage of military aircraft may differ significantly from the usage assumed during design. Likewise, individual aircraft within a force may experience a widely varied pattern of usage severity as compared to the average aircraft. Thus, inspection intervals, which are determined by predicting the amount of time the structure can safely sustain subcritical crack growth, must be continually adjusted for individual aircraft to ensure safety and to allow for modification and repair on a timely and economical basis.

Force management is the responsibility of the Air Force and is accomplished in accordance with the force management tasks of MIL-STD-1530A, Aircraft Structural Integrity Program (ASIP) [Reference 1], using a data package provided by the contractor for each new aircraft system. This data package consists of the necessary data acquisition and reduction techniques and analysis methods needed to acquire, evaluate, and utilize operational usage data in order to provide a continual update of in-service structural integrity.

A basic element of the force management data package is the individual aircraft tracking (IAT) program. The objective of the IAT program is to predict potential flaw growth in critical areas of each airframe based on individual aircraft usage data. A tracking analysis method is developed to establish and adjust inspection and repair intervals for each critical structural location of the airframe. This analysis provides the capability to predict crack growth rates, time to reach crack size limits, and crack length as a function of total flight time and usage. A data acquisition system is developed which is as simple as possible and is the minimum required to monitor those parameters necessary to support the tracking analysis method.

Current practice for acquisition of IAT usage data for fighter aircraft includes recording strain or center of gravity motion parameters (eg., normal load factor, n_z). The tracking analysis method then utilizes this data to estimate crack growth from assumed initial flaws in each critical point in the structure. Initial flaw size assumptions required for new aircraft are specified in Reference 2.

1. Background

The first IAT program for tracking crack growth in fighter aircraft was developed in conjunction with the F/RF-4 C/D and the F-4E(S) damage tolerance assessments (References 3-5). The present F-4 IAT program employs a counting accelerometer for data acquisition and a tracking analysis methodology which is termed the "damage index and equivalent S-N curve" system. Data acquisition is accomplished by recording normal load factor exceedances via counting accelerometers installed in each aircraft. The F-4 counting accelerometers are set to record n_z counts at 3, 4, 5, and 6 g's. Extrapolation techniques are used to determine n_z counts at

7 and 8 g's. In addition, VGH data (airspeed, load factor, altitude) are recorded on approximately thirteen percent of the force in order to provide background data for the IAT analysis.

The "damage and equivalent S-N curve" system (Reference 6) was developed for the F-4 to simplify the crack growth tracking process. Instead of conducting a cycle-by-cycle crack growth analysis for each critical location of each individual aircraft, only one number (the damage index) is computed for each aircraft based on individual usage. Through the damage index, crack growth at one location (the monitoring location) is determined. The amount of crack growth at other critical locations is evaluated by damage index limits that relate to the monitoring location. Individual flaw size assumptions used for all F-4 critical locations are based on the results of the previously mentioned damage tolerance assessments.

Equivalent S-N curves are used to convert individual aircraft counting accelerometer data to a damage index for each aircraft. These are not the standard S-N curves for fatigue which present stress versus number of cycles to failure for constant amplitude loading. These equivalent S-N curves represent flight-by-flight crack growth at the monitoring location and were developed from crack growth curves for three usages; mild, baseline, and severe (see Figure 1). To construct the equivalent S-N curves, crack growth testing was used to determine the percentage of total crack growth by each stress level in the flight-by-flight load history. Then, knowing the percent crack growth of each stress level and the number of cycles of each stress level at the operational limit and establishing the damage index at 1.0 at the

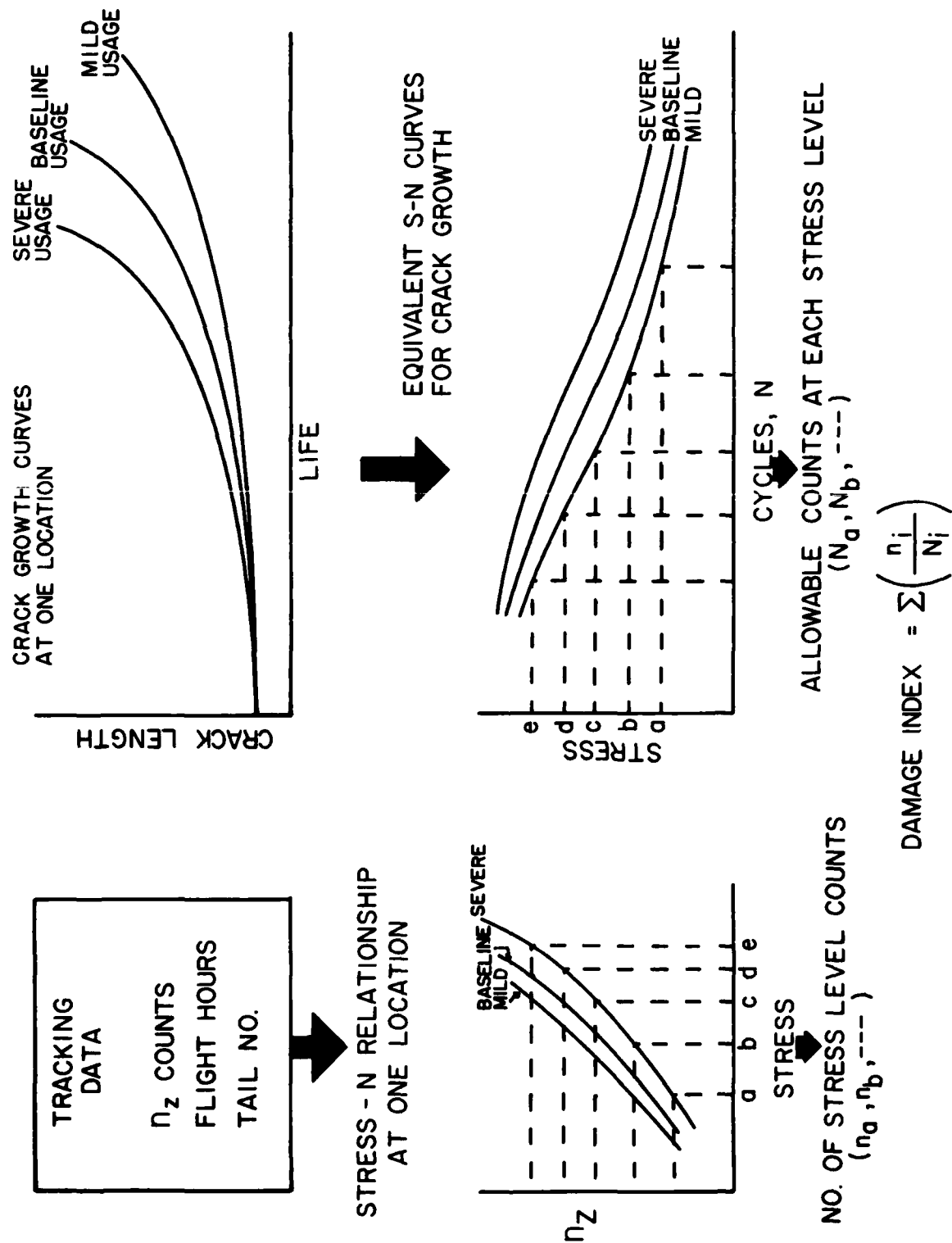


Figure 1. Schematic Representation of Current F-4 Tracking Analysis Method Employing the Damage Index and Equivalent S-N Curve System

operational limit, the allowable counts at each stress level were determined. Thus, the equivalent S-N curves show the number of cycles at each stress level necessary to reach the operational limit of the monitoring location (i.e., to obtain a damage index of 1.0).

Tracking data consisting of n_z counts, flight hours, and tail numbers are received from field operations on a periodic basis (normally monthly). Actual flight hours are not used directly in the structural life calculations but are used for other maintenance considerations involving avionics and engines. The n_z counts are examined and grouped into one of three usage categories according to severity. Then, using the known stress- n_z relationship for the monitoring location, the number of counts of cycles of each stress level are determined. Note that these stress level counts are those experienced by a particular aircraft in a particular time increment. These stress level counts are then divided by the allowable counts at each stress level and summed in a Miner's type analysis (Reference 7) to compute damage index for a particular aircraft.

For the F-4, the damage indices for all critical locations are based on n_z counts, airspeed, altitude, and gross weight. Relating the airspeed, altitude and gross weight of the aircraft to the number of load factor exceedances is a complex process and requires detailed analysis. Clearly, there is a need for a simpler and more direct method for tracking aircraft damage than the counting accelerometer method.

2. Crack Growth Gage Technique for IAT

A possible alternative IAT system which employs cracked metal coupons (i.e., crack growth gages) as the recording device was evaluated and is the central subject of this report. The approach consists of mounting a precracked

coupon onto a load-bearing structural member [References 8-14]. Theoretically, the coupon receives the same load excursions encountered by the structure (to within a predictable scaling factor) and responds with a measurable crack extension which may be related to the growth of another crack assumed to be present in a remote structural component. One may consider the cracked coupon as an analog computer which senses the load history, determines its effect on crack growth, and responds with measurable output (i.e., coupon crack extension).

Introducing an intentional flaw in a gage that is mounted on an aircraft would provide a direct method for assessing crack growth damage and for determining rates of crack growth as a function of usage. Using the crack growth gage as a tracking device would eliminate the gross assumptions associated with using the counting accelerometer (i.e., the assumed relationship between the values for airspeed, altitude, and gross weight and the number of n_z counts actually experienced). In addition, the crack growth gage would eliminate the need to go through the n_z counts analysis using Miner's rule to compute the aircraft damage index. Therefore, the damage index calculated by the crack growth gage method would be more accurate, more meaningful, and have less risk associated with it than the damage index computed by the counting accelerometer method.

The concept of the crack growth gage is shown schematically in Figure 2. The approach consists of employing linear elastic fracture mechanics analysis to relate the crack length measured in the gage (a_g) with the length of a real or assumed initial flaw located in the structure (a_s). The structural crack length is then related to the fraction of total aircraft life expended (N_i/N_f) in a normalized life scheme.

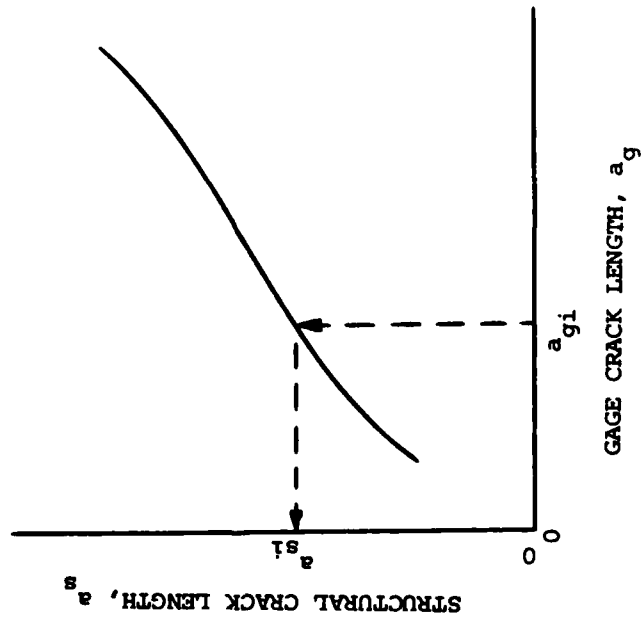
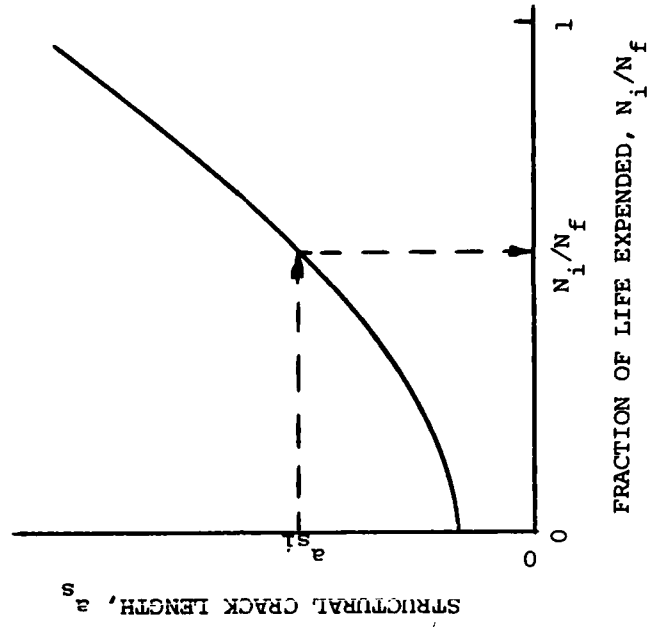


Figure 2. Schematic Representation of Damage Tracking Using the Crack Growth Gage Concept

A mathematical model (Reference 15) for relating the crack length in the coupon (crack growth gage) a_g to the growth of an assumed structural flaw a_s is shown schematically in Figure 3. The initial structural flaw size and shape is based on the appropriate design criteria (Reference 2), while the gage geometry may be selected for a given response. The ends of the crack growth gage are assumed to be fastened (eg., adhesively bonded, riveted, welded, etc.) to the structural member so that when the structural component is subjected to some remote stress (σ_s), an effective stress (σ_g) is transferred to the cracked gage. This relationship between structural and gage loads can be expressed in the form

$$\sigma_g = f\sigma_s \quad (1)$$

Here the load transfer function f may depend on geometry and material properties, but not on stress levels. Determining an expression for f is essentially a stress analysis problem which can be readily approached by several analytical and/or experimental techniques (References 8-10).

Now, assume that crack growth in the gage and structural materials can be described by a model of the form

$$\frac{da}{dN} = F(K) \quad (2)$$

Here da/dN is the fatigue crack growth rate and $F(K)$ is an appropriate function relating the stress intensity factor K , material properties, and other significant load variables. Much of the success of fracture mechanics techniques for analyzing crack growth problems lies in the fact that such crack growth models are readily available and are applicable for many structural materials. Solving Equation 2 for cyclic life N , and observing that at any instant of time the gage and structural defects receive the same number of load cycles leads to

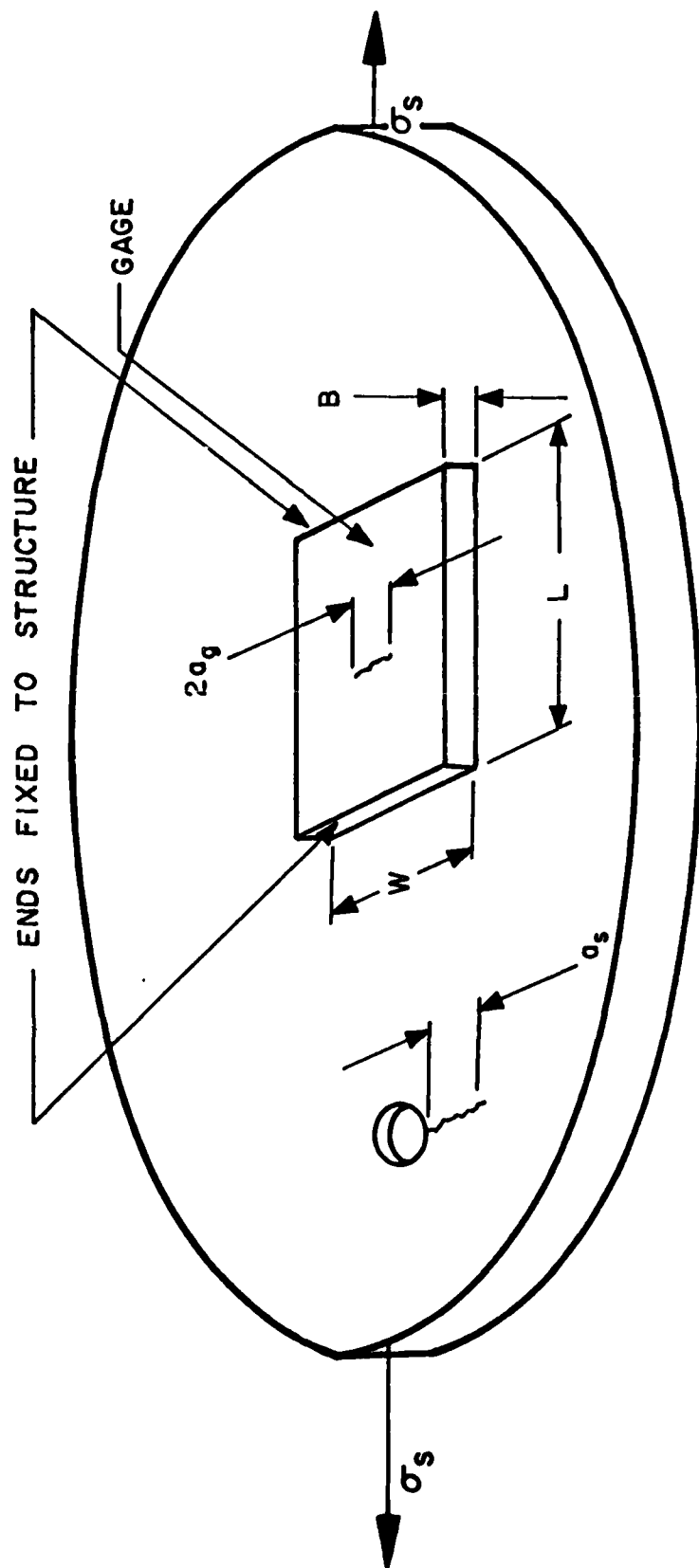


Figure 3. Schematic View of Crack Growth Gage Attached To Flawed Structural Component

$$N = \int_{a_{is}}^{a_s} \frac{da}{F_s(K)} = \int_{a_{ig}}^{a_s} \frac{da}{F_g(K)} \quad (3)$$

Here a_i and a are the initial and final crack lengths, while the subscripts s and g refer, respectively, to structural and gage quantities.

An interesting special case occurs when crack growth in the structural and gage materials can be described by the Paris law (Reference 16)

$$\frac{da}{dN} = C\bar{K}^m = F(K) \quad (4)$$

Here \bar{K} is the range in cyclic stress intensity factor and C and m are empirical constants. Now, expressing \bar{K} in the standard form

$$\bar{K} = \bar{\sigma} \beta \sqrt{\pi a} \quad (5)$$

where $\bar{\sigma}$ is the cyclic stress, β is the flaw geometry dependent stress intensity factor coefficient (References 17-19), and a is the crack length, and combining Equations 1, 3, 4, and 5 leads to

$$N = \int_{a_{is}}^{a_s} \frac{da}{C_s (\bar{\sigma}_s \beta_s \sqrt{\pi a})^{m_s}} = \int_{a_{ig}}^{a_s} \frac{da}{C_g (f \bar{\sigma}_s \beta_g \sqrt{\pi a})^{m_g}} \quad (6)$$

Note that a is the dummy variable of integration in Equation 6 and that, while f and β depend on geometries and possibly material properties, neither function depends on the load level $\bar{\sigma}_s$.

Further assuming that the gage and structural materials have the same crack growth exponent $m_s = m_g = m$ (a reasonable assumption if gage and structure are made from the same material) leads to

$$\int_{a_{is}}^{a_s} \frac{da}{C_s (\beta_s \sqrt{\pi a})^m} = \int_{a_{ig}}^{a_g} \frac{da}{C_g (f \beta_g \sqrt{\pi a})^m} \quad (7)$$

Note that all stress level terms effectively cancel in Equation 7. Although the expression no longer specifies the cyclic life N , it still represents a valid relationship between gage and structural quantities. The material properties, C_s , C_g , and m can be determined from conventional baseline testing, the stress intensity factor coefficients, β , are readily available from handbooks (References 17-19) or are obtainable by standard analysis methods, and the initial gage and structural crack lengths a_{ig} and a_{is} are specified. Equation 7 can then be integrated numerically to obtain the structural crack size a_s as a function of gage crack size a_g . Thus, measuring the gage crack length determines the growth of the initially assumed structural defect during service.

3. Program Objective

The objective of this program was to determine the feasibility of the crack growth gage as a method for monitoring potential crack growth damage in fatigue critical areas of F-4 C/D aircraft structure. Testing was divided into three tasks. The major task consisted of mounting crack growth gages to a full-scale F-4 C/D test article and collecting crack growth data from the gages at specified intervals. The second task was comprised of conducting gage qualification tests in accordance with MIL-STD-810C (Environmental Test Methods) requirements. This military

standard establishes uniform environmental test methods for determining the resistance of equipment to the effects of natural and induced environments peculiar to military operations. (Inconsistencies in constant amplitude test results prevented the MIL-STD-810C qualification tests from being started.) The third task was to determine an appropriate method for collecting data from the gage.

SECTION II

TEST PROGRAM

1. Introduction

The purpose of the test program was to obtain the experimental data necessary to characterize and validate the behavior of the crack growth gage. The testing was divided into two phases. Phase I consisted of laboratory testing required to determine gage response and predictability when the gage was mounted on a carrier specimen and subjected to constant amplitude and spectrum load conditions. Phase II consisted of testing crack growth gages attached to the F-4 C/D full-scale fatigue test article located in the Structural Test Branch (FBT) facility of the Air Force Flight Dynamics Laboratory (AFFDL).

2. Test Materials

2.1 Alloy Selection

The material selected for both the crack growth gage and the carrier specimen was 7075-T651 aluminum. This material has minimal lot-to-lot variability, is readily obtainable from vendor stores, and has readily available da/dN , K_{IC} , and standard mechanical data required for crack growth analysis. 7075-T651 was selected because of its wide usage in aircraft components, including the F-4 wing skin.

2.2 da/dN Coupons

These test coupons were fabricated from the same material as the crack growth gages. They consisted of 0.08 inch thick end sections and a 0.04 inch thick neck-down section. The neck-down test section was 1.50 inches wide by 4.75 inches long. The initial flaw was a 0.050 inch diameter hole with a 0.025 inch electric discharge machined (EDM) notch on both sides of the hole so that the total starter flaw was 0.100 inch in length. See Figure 4 for details.

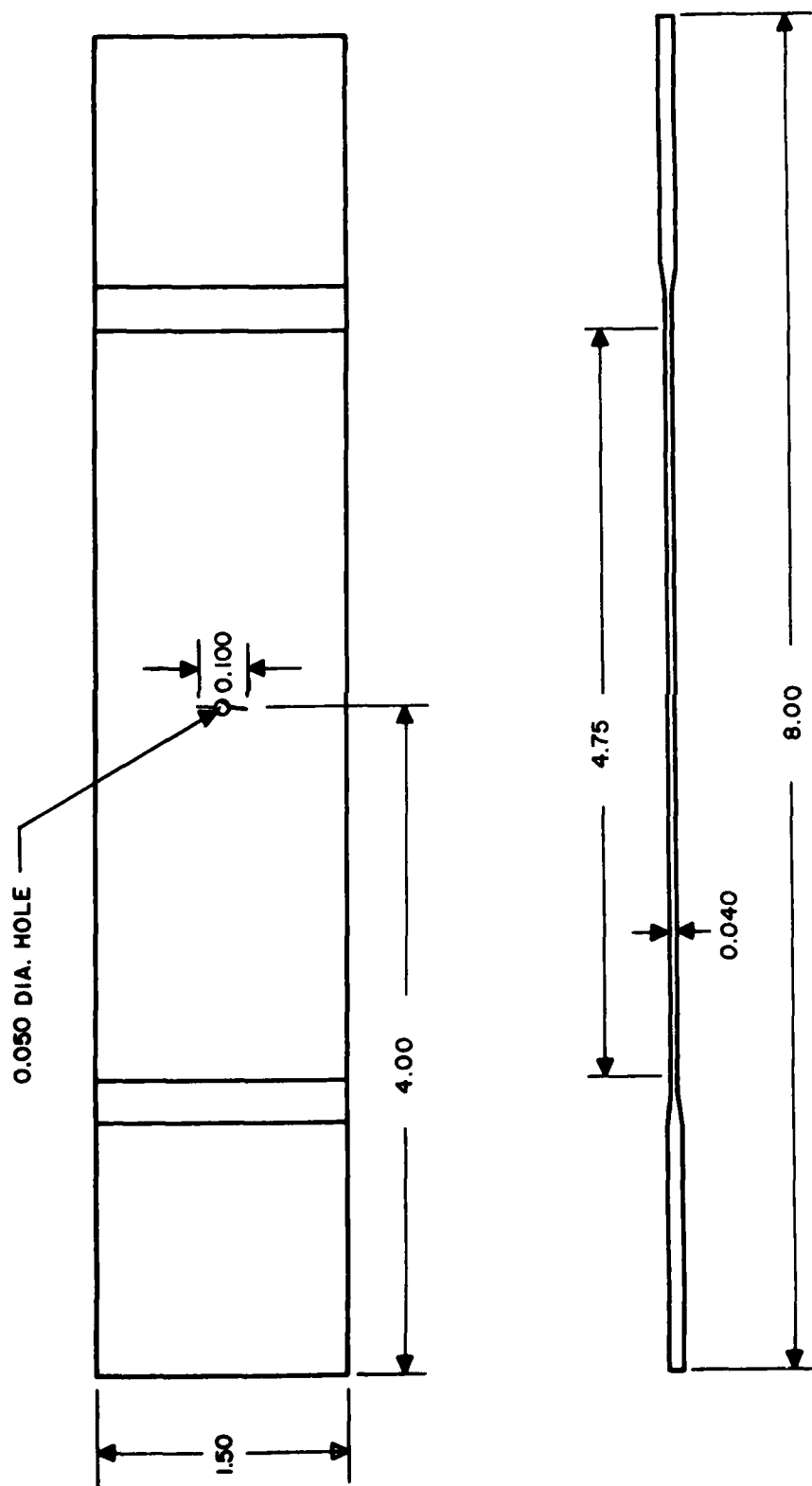


Figure 4. da/dN Coupon

2.3 Crack Growth Gage Design

The crack growth gage for this program was designed by McDonnell Aircraft Company (McAir) specifically for application to the lower wing skin of the F-4 (Reference 14). The design was based upon the following criteria: (1) the gage must give measurable crack growth for each 1000 spectrum hours of test life, (2) the gage must be durably bonded to the aircraft, and (3) the gage must not buckle under the maximum compressive stress in the spectrum.

The objective in selecting gage dimensions was to create the smallest gage that would produce (1) adequate crack growth to permit measurement with simple equipment and (2) good load transfer through the adhesive and the gage. The gage as dimensioned in Figure 5 was designed to produce approximately one inch of crack growth in 12000 spectrum hours, an average of 0.09 inch growth for each ten percent of the gage life. The configuration as shown in Figure 5 has a 0.100 inch starter slot created by drilling a 0.050 inch diameter hole, then 0.025 inch EDM notches are cut on each side of the hole.

2.4 Adhesive Selection

American Cyanamid's FM-73 was selected for this program. It was chosen over the other state-of-the art epoxy film adhesives as having the best combination of strength, temperature resistance, environmental durability, and superiority for in-the-field bonding. FM-73 was demonstrated in the Primary Adhesively Bonded Structure Technology (PABST) program (Reference 20) as a feasible adhesive for bonding aluminum aircraft structure.

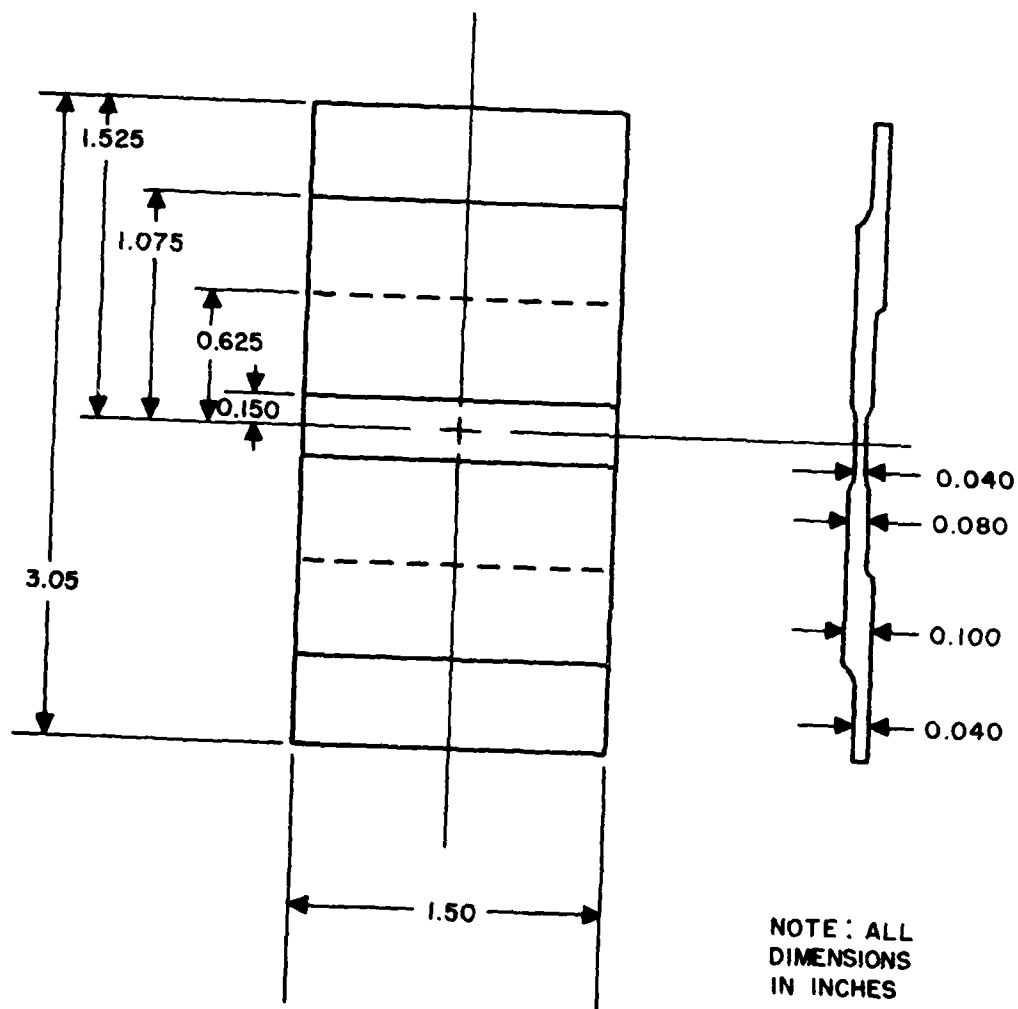


Figure 5. Crack Growth Gage

3. Test Procedures

3.1 Crack Growth Gage Bonding Technique

The following list describes the essential steps used to prepare the carrier specimens and the procedure used to bond the crack growth gages to the carrier specimens.

1. Carrier Specimen Preparation

- a. Sand blast surface to be bonded.
- b. Clean surface with soap and water and wipe dry.
- c. Etch surface with M-Prep Conditioner A (a water based acidic surface cleaner).
- d. Rinse with clear water and air dry.
- e. Wipe with MEK.







2. Bonding Procedures

- a. Cut cold FM-73 sheet to size and insert between parts.
- b. Place carrier specimen-crack growth gage combination (Figure 6) in oven.
- c. Raise temperature of oven so thermocouple alongside carrier specimen measures 255°F.
- d. Bond at 255°F for one hour.
- e. Oven cool to room temperature.

3.2 Crack Monitoring

Crack growth was monitored either by visual observation using stereo zoom microscopes and ruled scales or by using Fax-Film.

Fax-Film, a registered trade name of the Clevite Corporation, is a facsimile film which has the unique ability to produce a replica of the surface to which it has been applied. This ability has a tremendous

-  LEAD WEIGHT  0.0625 & 0.125 INCH RUBBER
 0.25 INCH ALUMINUM PLATE  CRACK GROWTH GAGE
 0.75 INCH RUBBER SPONGE  0.34 INCH CARRIER SPECIMEN

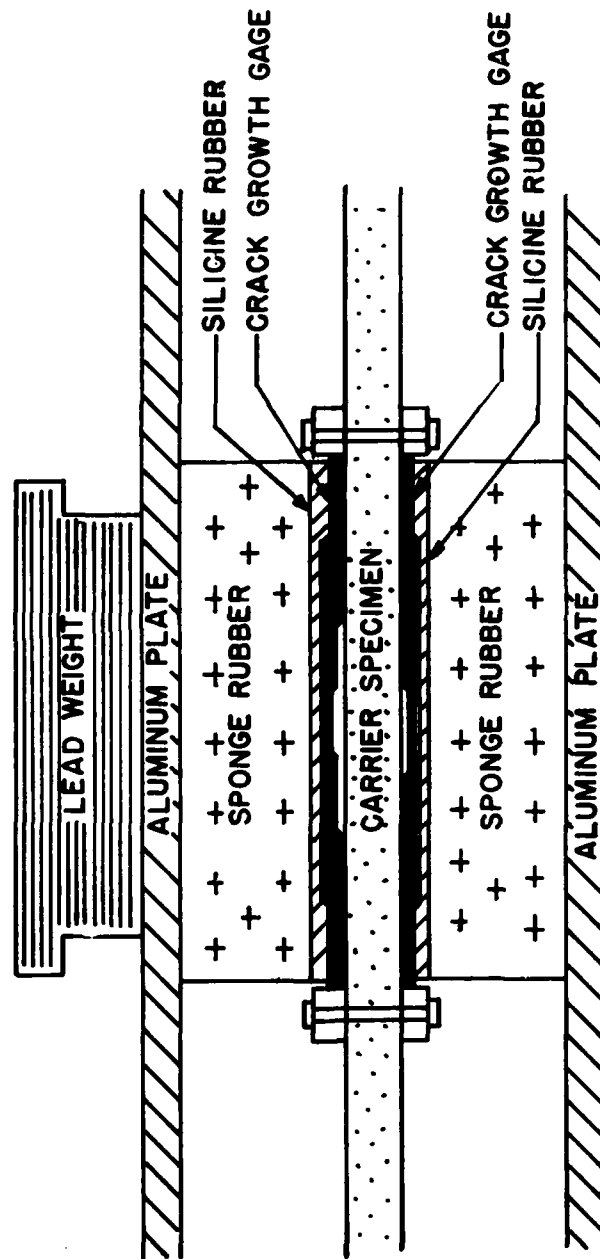


Figure 6. Crack Growth Gage Bonding Technique

advantage over the human eye. Fax-Film replicas can be magnified to several hundred power, microscopically studied, photographed, viewed by many people at the same time with the aid of the slide projector, and stored as a permanent record.

The materials needed to produce a Fax-Film replica include:

1. Fax-Film (cellulose acetate)
2. Film holder (2 inch x 2 inch slide mount)
3. Solvent (usually acetone)
4. Cleaning materials (cotton, applicators)

The following list describes the steps needed to obtain replicas using Fax-Film.

1. Clean surface thoroughly with acetone and cotton. Remove all grease, dirt, and lint from the surface.
2. Cut Fax-Film to size larger than the area to be inspected. Care should be exercised in keeping all foreign matter, finger prints, and scratches from the surface of the film.
3. Moisten either the film or the surface to be inspected with acetone and place the film on the surface. Avoid air bubbles and prevent any lateral movement or sliding of the film.
4. Hold film securely with constant pressure for approximately one minute. This time may vary according to the amount of acetone used.
5. Peel the replica from the surface and immediately place in the film holder.

The replica is now ready to be viewed in a microscope or to be projected through a lens system onto a screen. Figure 7 represents Fax-Film replica of the five precracked crack growth gages which were bonded to the lower wing skin of the full-scale F-4 C/D fatigue test article.

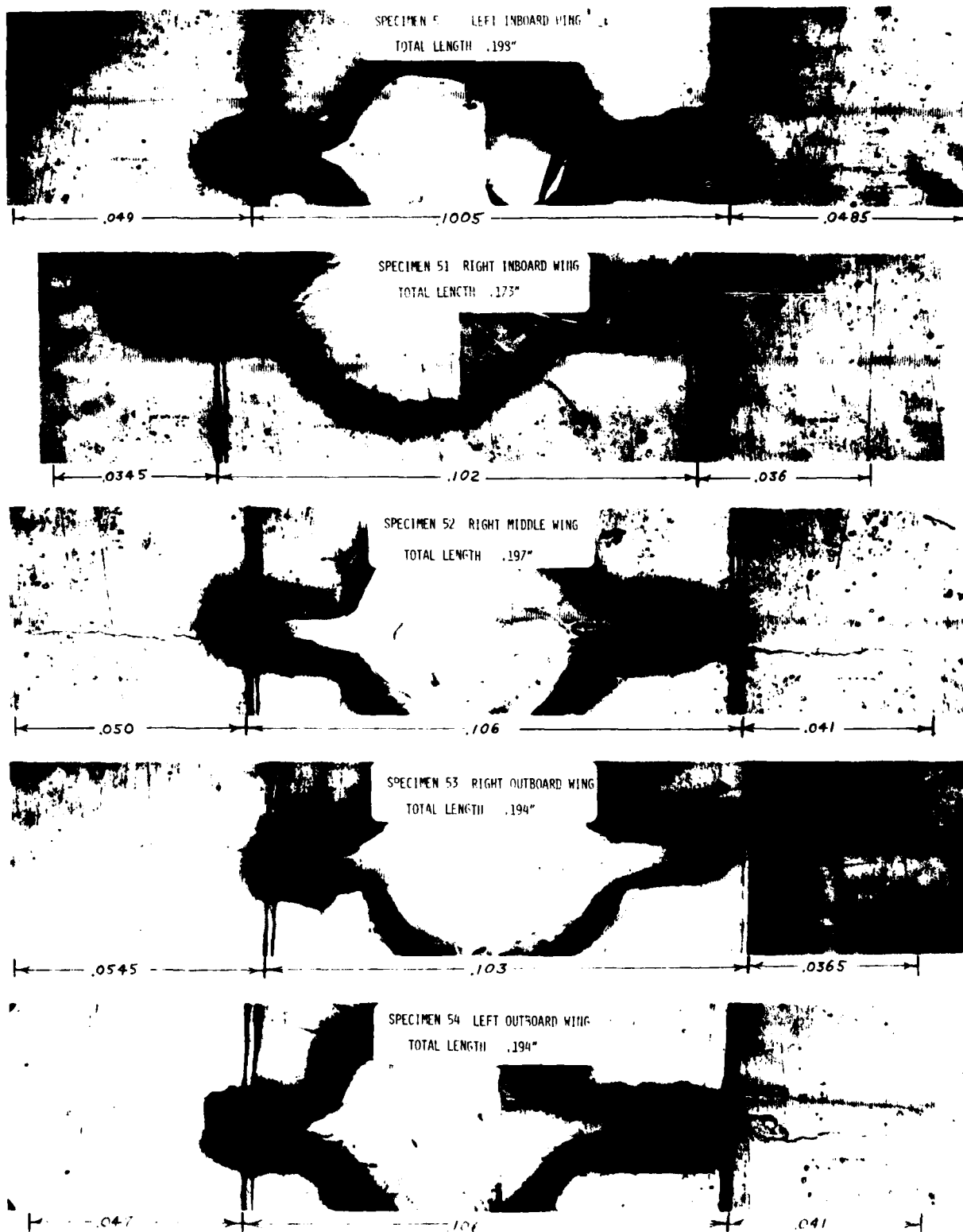


Figure 7. Example of Fax-Film Replication (Magnification 70X)

The accuracy of both systems, stereo zoom microscope and Fax-Film, is plus or minus 0.002 inch.

3.3 Precracking of Crack Growth Gages

The crack growth gages that were bonded to the lower wing skin of the F-4 full-scale fatigue test article and those used early in the test program were precracked by clamping them back-to-back with a non-slotted crack growth gage and fatigue cycling them at 10-12 ksi at a stress ratio of zero. Precracking was halted when crack lengths on both sides of the slot reached a nominal 0.050 inch. The configuration shown in Figure 8 was used when the gages were precracked. Two inches of length were cut from both sides of the gage after the precracking procedure was complete.

During testing it was determined that precracking was not required for crack growth gages which were to be bonded to carrier specimens. This determination was a result of a comparison of test data using precracked and non-precracked gages.

3.4 Primary Specimen Testing

All specimens were tested in analog controlled hydraulically driven servo-valve test machines (MTS Systems). Specimens were clamped by hydraulic powered grips.

3.5 da/dN Coupon Testing

Testing consisted of applying constant amplitude loading at 10-12 ksi and periodically recording crack length and cycles. Testing was conducted in a 20 KIP MTS machine utilizing a 20 KIP capacity load cell. The cyclic rate used was 2.5 Hz. Appendix A contains detailed results of the da/dN coupon tests.

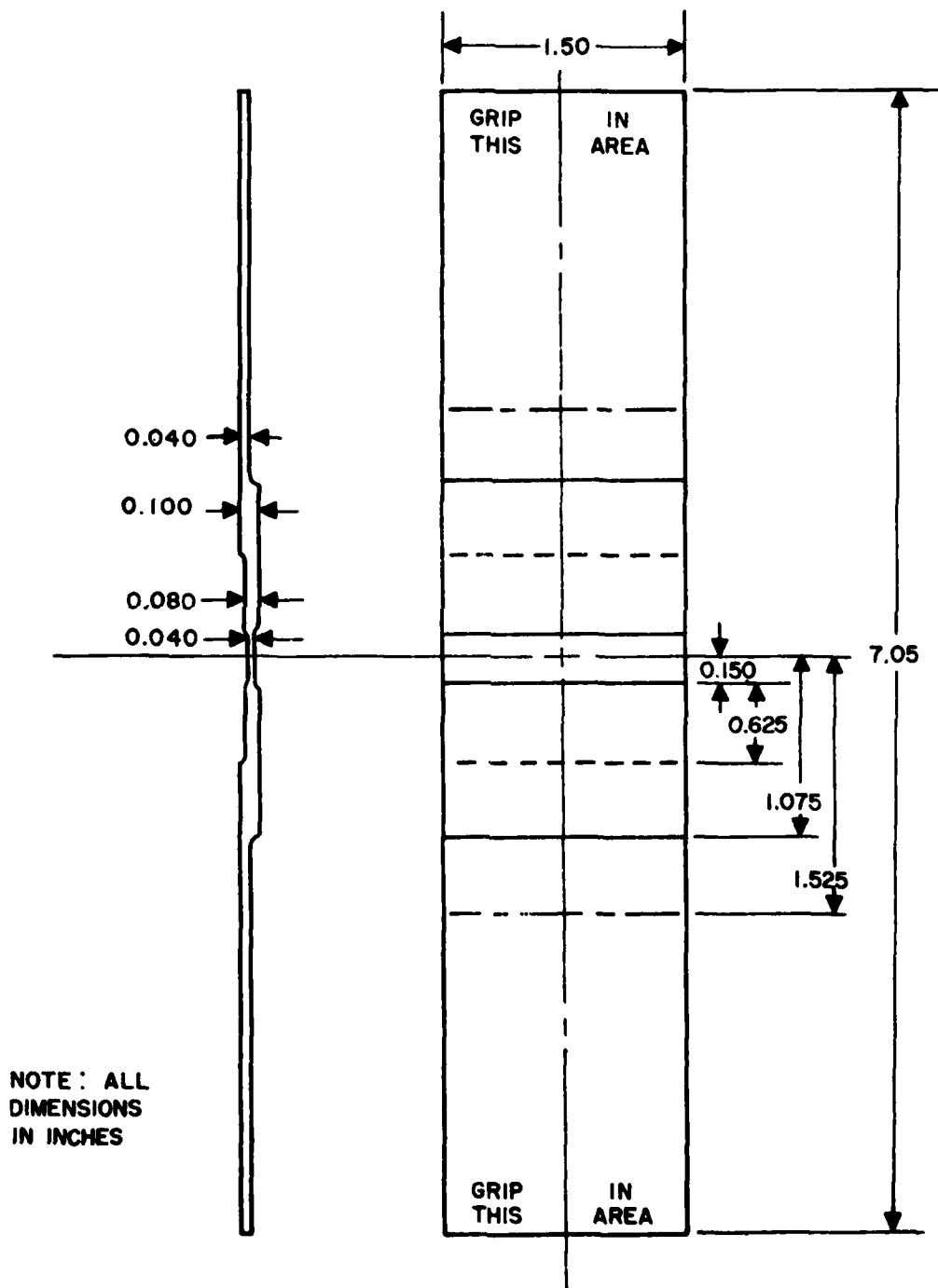


Figure 8. Crack Growth Gage Precracking Configuration

3.6 Constant Amplitude Testing

Constant amplitude tests on carrier specimens with attached crack growth gages were run at a stress ratio of zero with a maximum stress of 21.0 ksi. Testing was conducted in a 100 KIP MTS machine utilizing a 100 KIP capacity load cell. The cyclic rate used was 0.5 Hz. Three gage configurations were tested: (1) normal length gage - bonded, (2) full length gage (precracking length) - bonded, and (3) full length gage (precracking length) - bonded and bolted. The three configurations are shown in Figures 9, 10 and 11.

3.7 Strain Gage Instrumentation

Selected carrier specimens and crack growth gages were strain gaged with Micro-Measurement foil-type miniature strain gages to measure load transferred through the crack growth gage. The strain gages and the applied loads were monitored through the AFFDL-FBT Data Acquisition System. This system uses multi-channel high-speed A-to-D multiplexers output to PDP-11 minicomputers which are linked to a SEL-86 computer. Data sampling rates were as high as 50,000 samples per second. The strain surveys were performed under static load conditions. Appendix B contains detailed drawings of strain gage locations and tabulated data resulting from the strain gage measurements.

4. Fatigue Test Article

An on-going fatigue test of a full-scale F-4 C/D test article provided a convenient test bed for evaluating the crack growth gage concept for use with actual aircraft. The purpose of the F-4 C/D full-scale fatigue test is to provide full-scale test verification of several

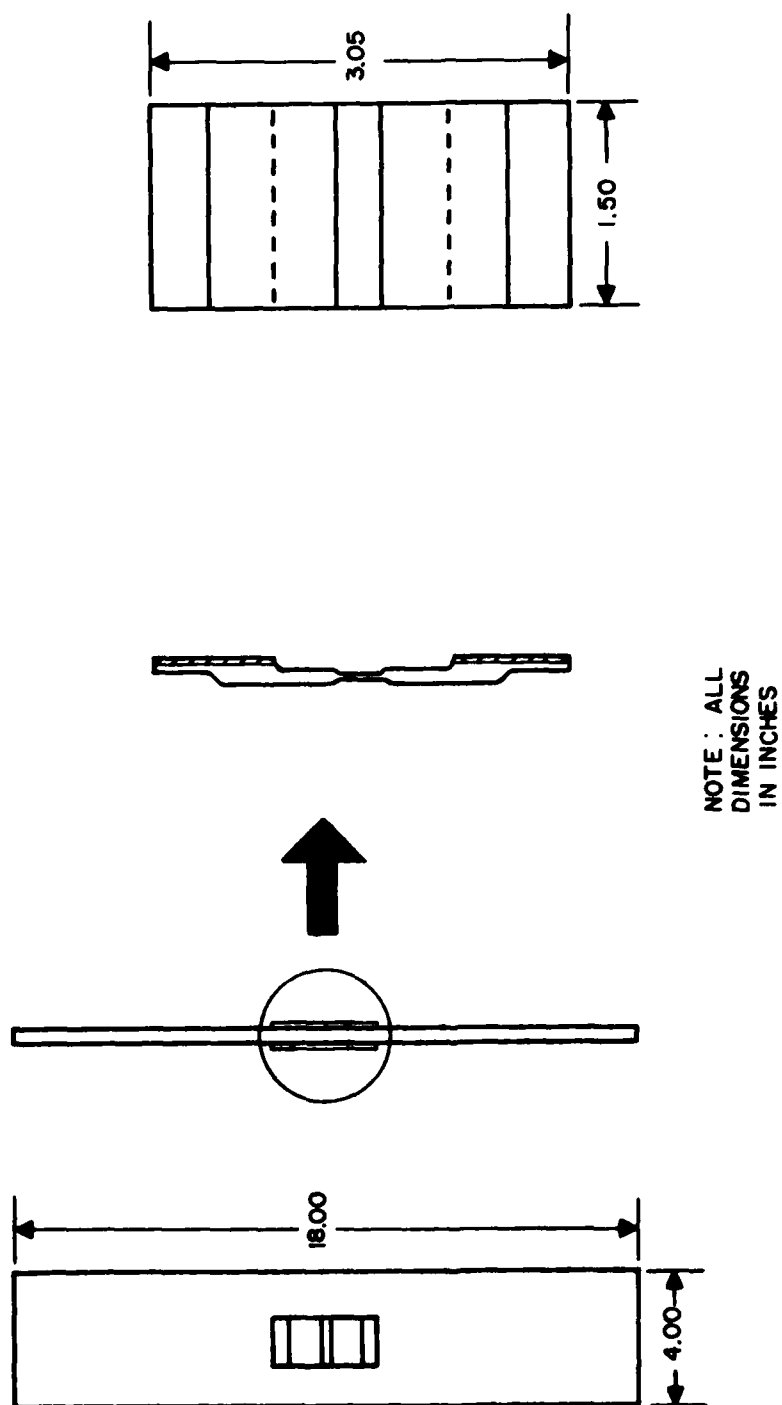
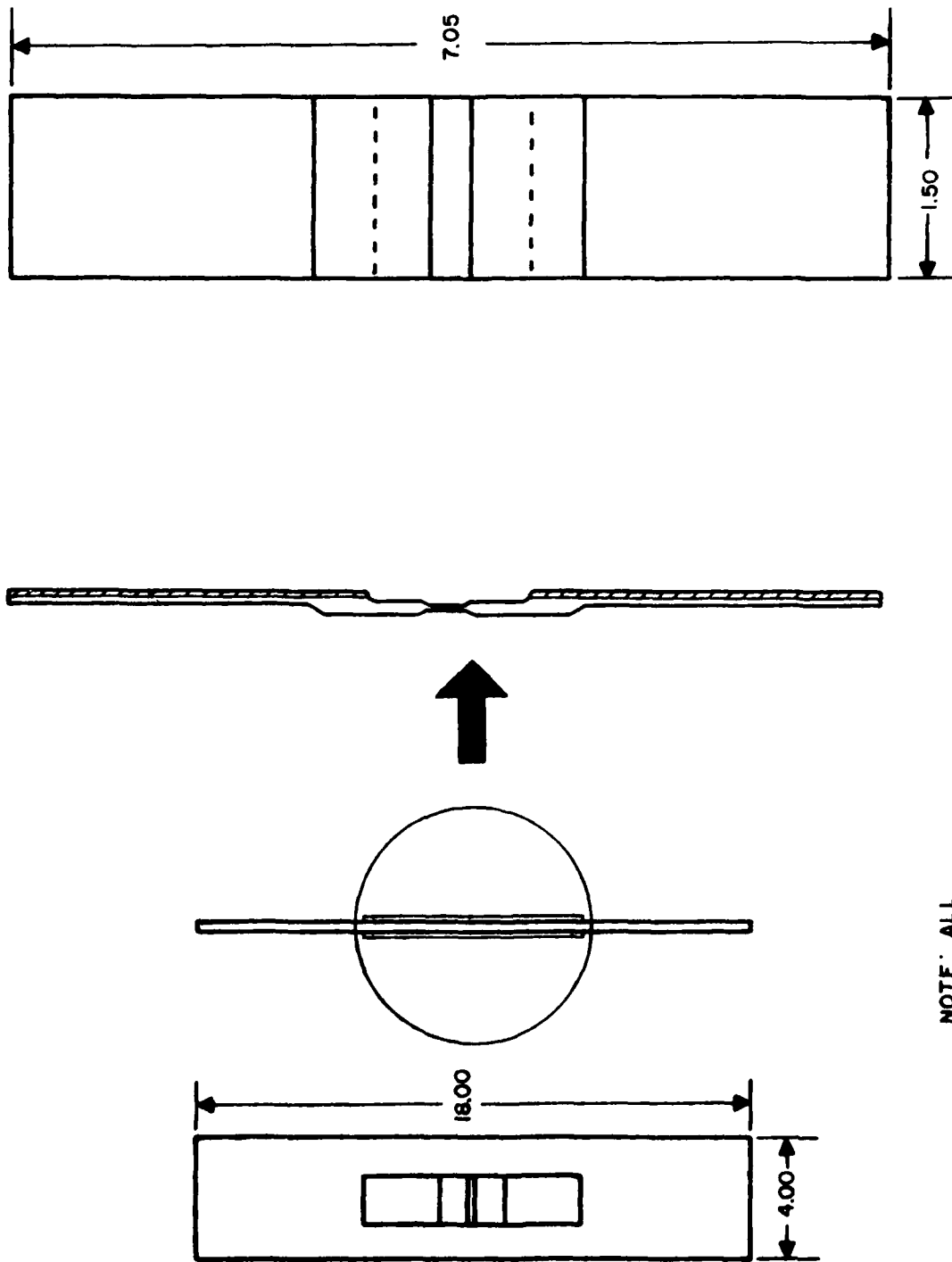


Figure 9. Crack Growth Gage Test Configuration: Normal Length Gage - Bonded



NOTE: ALL
DIMENSIONS
IN INCHES

Figure 10. Crack Growth Gage Test Configuration: Full Length Gage (With
Precracking Tabs) - Bonded

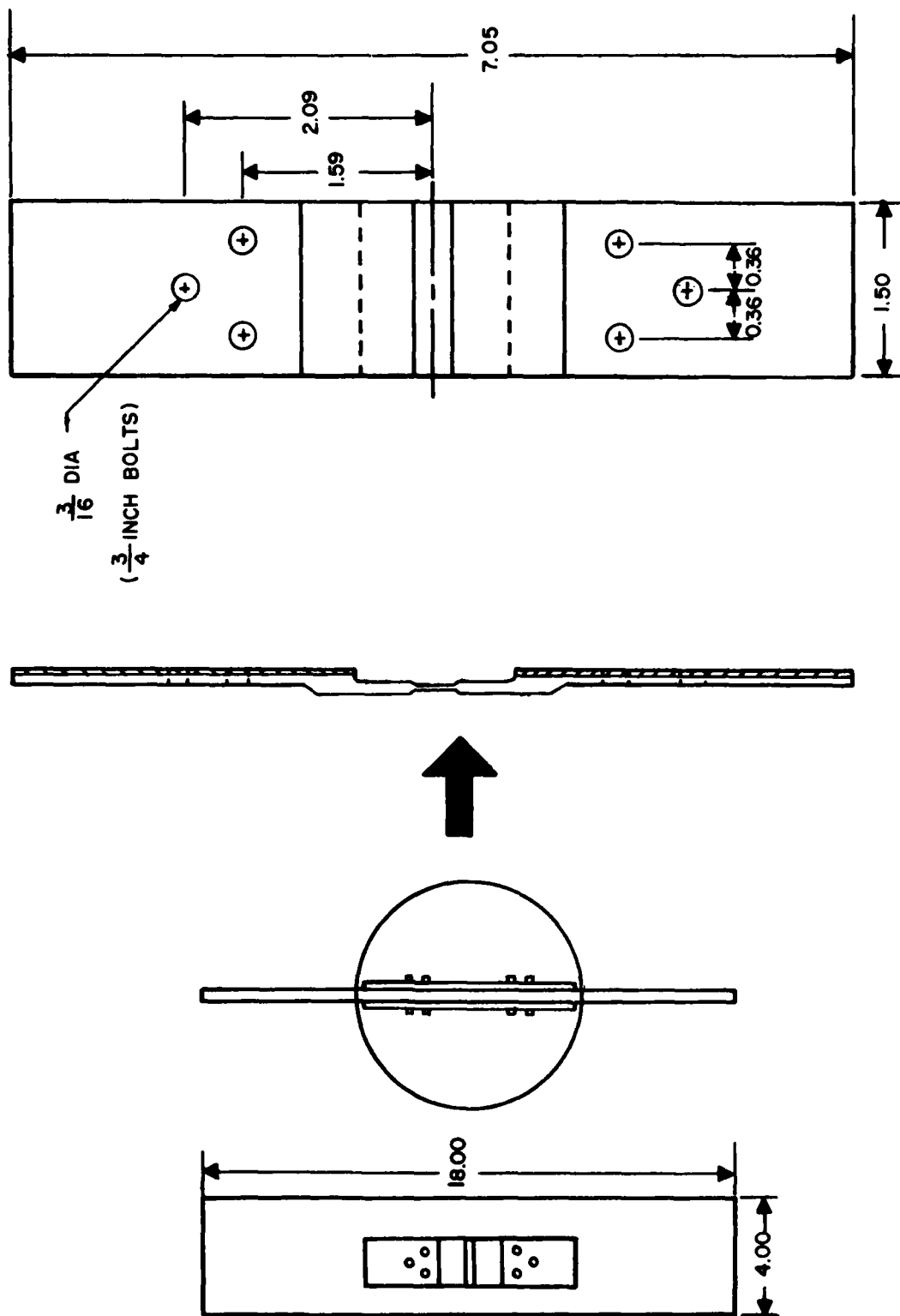


Figure 11. Crack Growth Gage Test Configuration: Full Length Gage (With Precracking Tabs) - Bonded and Bolted

life extension modifications including those designed to extend the life to 8000 flight hours of F-4 ASIP baseline usage. At the equivalent of 4000 flight hours of baseline usage, the full-scale fatigue test was stopped temporarily to implement the modifications mentioned above. Thus, the test was in a hold status and provided an excellent opportunity to attach crack growth gages to the test structure.

A contract was established with McAir to conduct detailed analysis and testing which would evaluate the ability of the crack growth gage concept to monitor potential crack growth damage in fatigue critical areas of the F-4 C/D aircraft structure. Under the contract, McAir selected three external locations on the lower wing skin for monitoring wing fatigue critical regions. Gage application sites were based on gage configuration, predicted behavior, bonding procedure, and the following criteria: (1) sites should be near fracture critical areas, (2) sites should experience about 30 ksi limit stress level, (3) sites should avoid high stress gradients, fastener patterns, taper-loks, and load pads.

The locations chosen for attaching the crack growth gages to the lower wing skin of the right wing are shown in Figure 12. These locations were chosen because they are at or near control points for which crack growth damage is calculated in the present F-4 IAT program. Also, stress spectra were already developed and crack growth analysis and test data were available from the previous F-4 damage tolerance assessments.

Site 1 is an area of moderately high design limit stress (see design limit stress contours for the lower wing skin in Figure 13), and cracks have been found in this area in previous full-scale fatigue tests. Site 2 is located in an area near Butt Line (B. L.) 100 which has a slightly

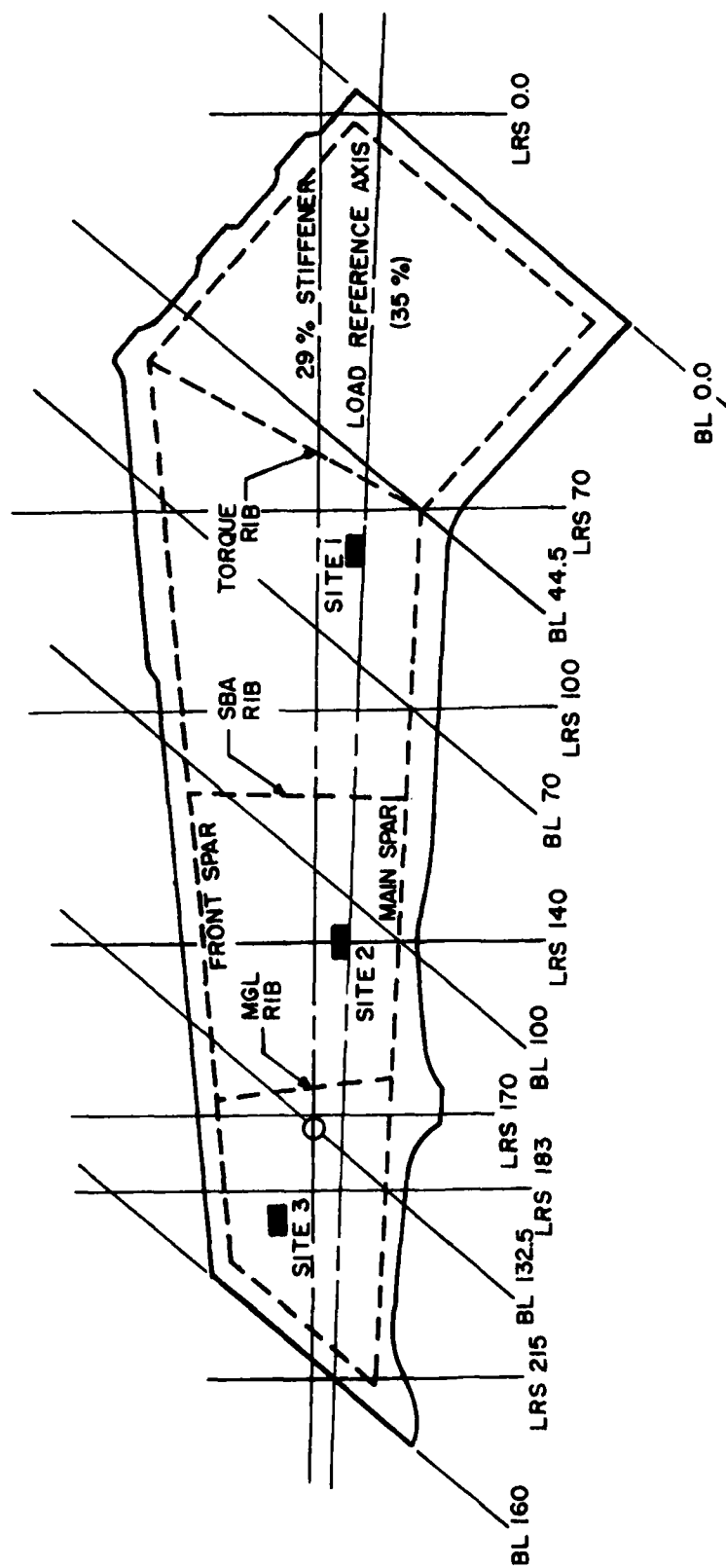


Figure 12. Selected Gage Sites

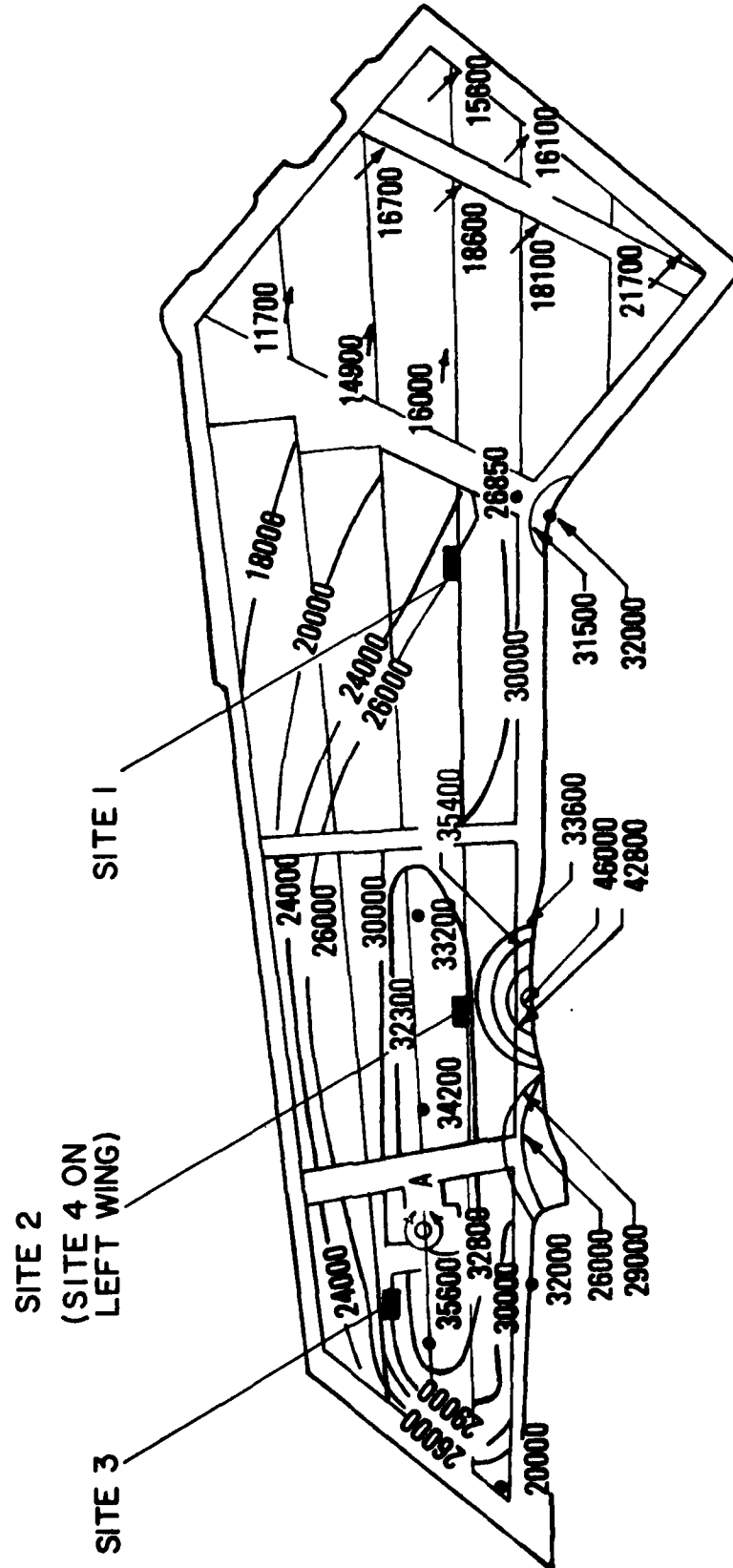


Figure 13. Relationship of Selected Gage Sites to Stress Contours on Lower Wing Skin

higher stress level (between 32.3 ksi and 33.6 ksi at limit load) than the other gage sites. Site 3 is located near the pylon hole. This is also an area of moderately high stresses, and cracks have been found in this area in operational aircraft during service as well as in previous full-scale fatigue tests. A fourth gage was installed at the duplicate location of Site 2 on the left wing.

4.1 Fatigue Test Article Surface Preparation

The surface of the lower wing skin where the gages were attached was prepared in the following manner. The preparation consisted of abrasion, followed by solvent wiping, followed by Pasa Jell 105 treatment, and finally by a decomped water rinse. This treatment, standard for field repair, was performed by McAir personnel.

The crack growth gages were treated with sulfuric acid and sodium dichromite. In addition, the gages were primed with a corrosion inhibiting primer (BR-127) before attachment.

4.2 Bonding Process

The technique used in bonding the gages to the lower wing is shown in Figure 14. The FM-73 film adhesive was sandwiched between the crack growth gage and the wing skin. A thermocouple was affixed to the wing skin, and glass breather material and a vacuum bag were applied. A heating blanket was placed over the vacuum bag, and the bond area was heated to the cure temperature of the adhesive (250°F). The cure cycle involved a half-hour heat-up to the control temperature (300°F), one to 1.5 hours at temperature, followed by a half-hour cool down.

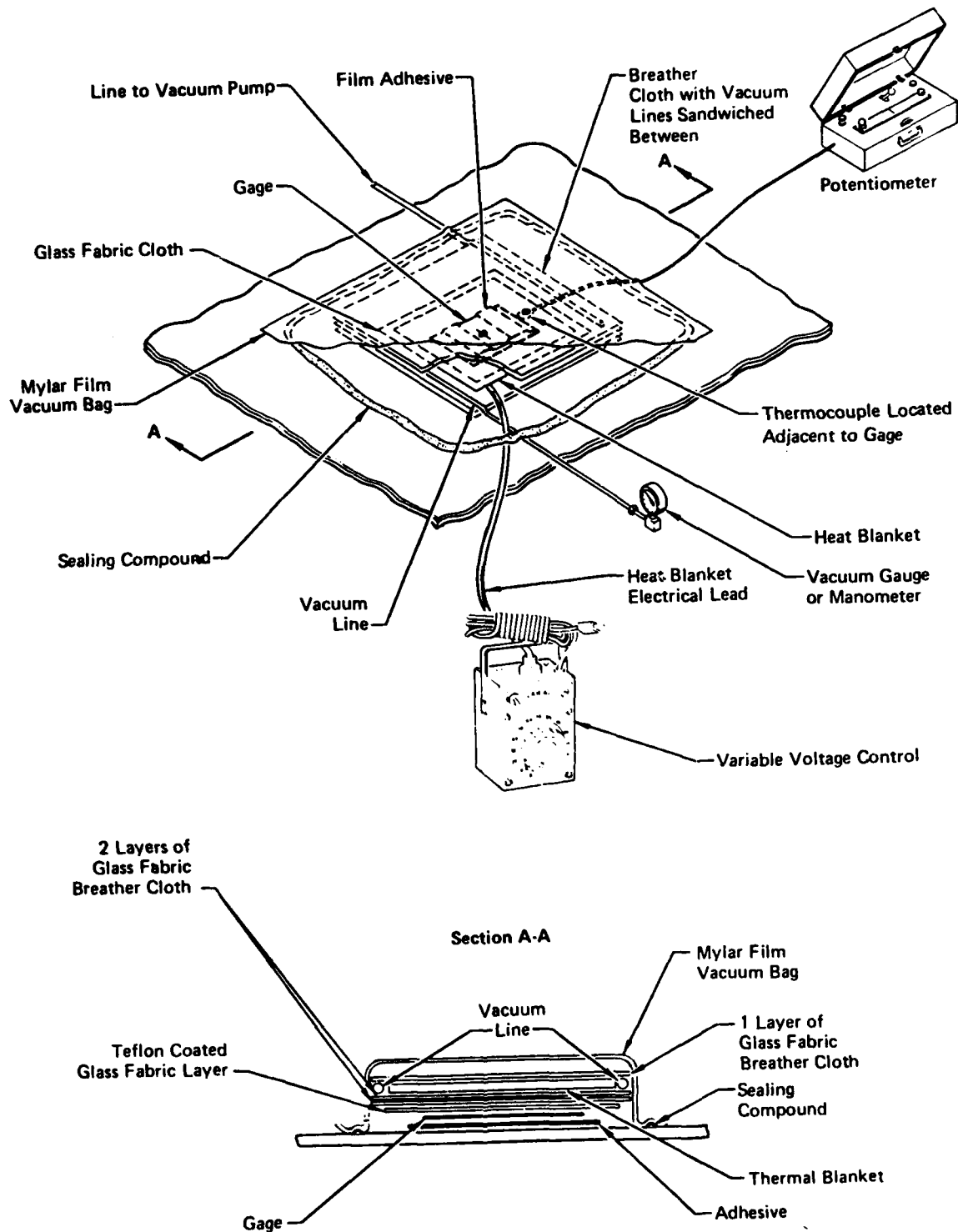


Figure 14. Vacuum Bag Technique of Bonding Crack Growth Gages

The elevated skin temperature associated with the bonding procedure required special precautions and controls to prevent residual stress relaxation near taper-loks and cold-worked holes. Gage bonding was performed so that temperatures near such areas were held to a maximum of 200°F. This was accomplished by locating the crack growth gages at least 2.5 inches from the nearest fastener pattern. In addition, thermocouples at the nearest fastener pattern were monitored during bonding and the temperature was held to a maximum of 200°F.

SECTION III

PRESENTATION AND DISCUSSION OF EXPERIMENTAL RESULTS

The purpose of the experimental test program was to obtain data necessary to verify the performance and predictability of the crack growth gage. The constant amplitude tests and the fatigue test article effort are described in detail in the following sections.

1. Constant Amplitude Test Results

All tests performed in this phase of the test program were intended to determine the crack growth behavior of the crack growth gage. Gages were bonded with FM-73 adhesive to carrier specimens and tested under constant amplitude conditions (21.0 ksi) in order to determine gage performance and predictability.

Four constant amplitude tests (i.e., two crack growth gages per carrier specimen) were completed and results showed two areas for concern. Figure 15 is a plot of the first eight crack growth gage tests compared with the results of tests performed by McAir. Scatter in the McAir tests was minimal while tests performed by AFFDL had a large amount of scatter. Also note that the crack growth rate was much slower for the AFFDL gages when compared with the McAir tested gages.

Gages 8A/B, 30A/B, and 24A/B were precracked prior to being bonded on the carrier specimens. Gages 18A/B were not precracked. All the gages tested by McAir were precracked to a 2a of approximately 0.2 inch. Since gages that were installed on the F-4 fatigue article were precracked to a 2a of 0.2 inch, this length was used as an initial starting point for all

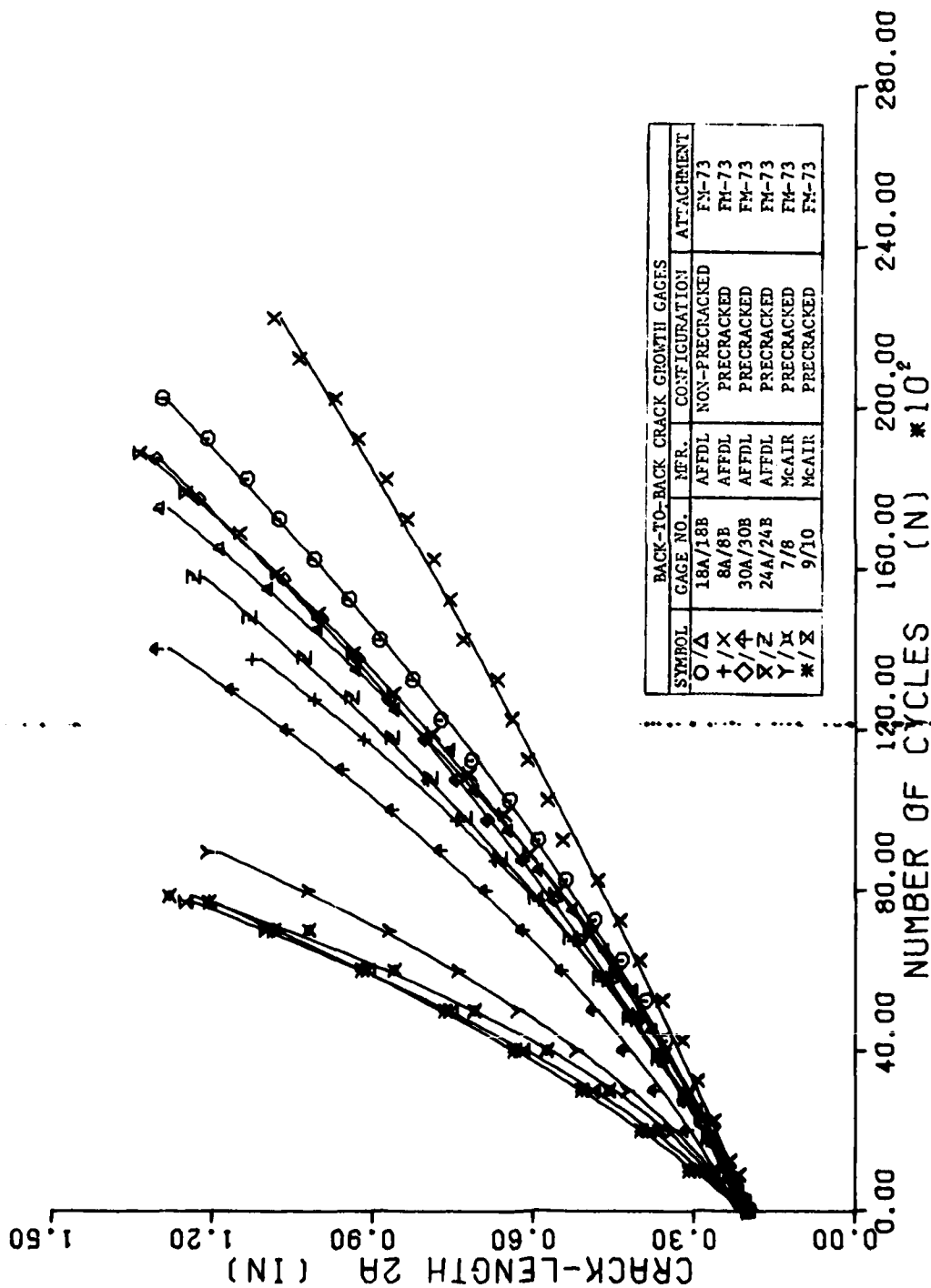


Figure 15. Constant Amplitude Tests - AFFDL Tests of AFFDL Gages Compared with McAir Tests of McAir Gages

constant amplitude test results. The number of cycles used to reach a $2a$ of 0.2 inch was not considered.

Differences in procedures used by McAir and AFFDL were investigated in an attempt to explain the variations in the test results. It was discovered that gages manufactured for the AFFDL were machined as opposed to chem-milled McAir gages. Therefore, the possibility of residual stresses existed in AFFDL manufactured gages that could have caused non-uniform crack growth through the gage thickness. McAir also installed a 0.020 inch thick teflon pad under the unbonded section of the gage to help prevent the adhesive from entering the cracked portion of the gage and also to restrain out of plane deformations. No such pad was used by AFFDL. During the bonding process, McAir used a vacuum bag to apply the necessary pressure for attaching the gage to the carrier specimen. AFFDL used lead weights to provide the required load.

To investigate the possibility of residual stresses in the AFFDL manufactured gages, four chem-milled gages were obtained from McAir and were bonded and tested by AFFDL personnel. Results of these tests are shown in Figure 16 along with the results of McAir previous tests. It can be seen in Figure 16 that the results of the AFFDL tests with McAir gages fall between the original results of McAir and the results obtained by the AFFDL with the machined gages. Therefore, the possibility of both the residual stresses in the AFFDL machined gages and improper bonding procedures by AFFDL personnel remained as a possible explanation for the inconsistent results.

The next tests consisted of using full length gages (i.e., gage with precracking tabs attached). In the first test a full length gage was bonded

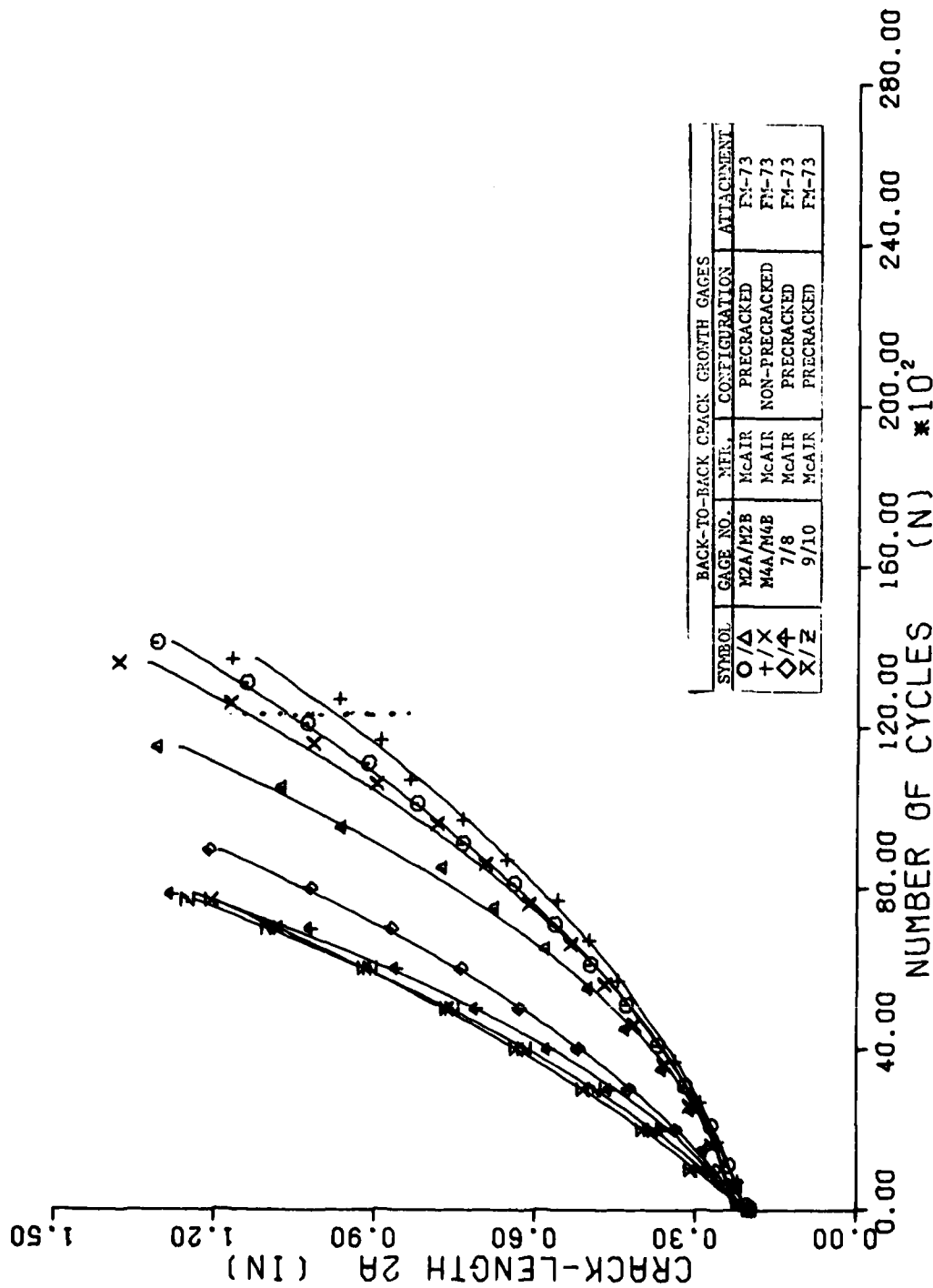


Figure 16. Constant Amplitude Tests - AFFDL Tests of McAir Gages Compared with McAir Tests of McAir Gages

with FM-73 (Figure 9) and a constant amplitude test was performed. The second test consisted of bonding and bolting full length gages on a carrier specimen (Figure 10). The objective of using the bolts was to determine if this method of gage attachment would eliminate (1) the variation between McAir and AFFDL test results and (2) the scatter in the AFFDL test.

Figure 17 shows the results of these tests compared with the McAir curves. Both gages that were bolted on showed very consistent results. One of the non-bolted full length gage tests showed the same trend as the two bolted gages. However, the test results for the other non-bolted full length gage fell approximately half-way between the other full length gage tests and the McAir curves. No explanation could be found for the latter results.

Since residual stresses were suspected in the AFFDL machined gages, the machining procedure was modified to include smaller cuts during manufacture of the gages and tighter specifications for the final product. One set of these new gages was tested. The results of this test (Figure 18) show that the scatter between the gages was reduced but the crack growth rate still remained less than the crack growth rate in the McAir tests.

As a result of these tests, McAir performed a constant amplitude test on gages that were bonded by AFFDL personnel. The results of this test are shown in Figure 19. These curves fall well within the AFFDL scatter band of previous tests; therefore, both the theories on residual stress in the AFFDL machined gages and the AFFDL bonding procedure were still suspect.

A test matrix by gage number is shown in Table 1. A summary of the test program and its results is shown in a flow diagram in Table 2.

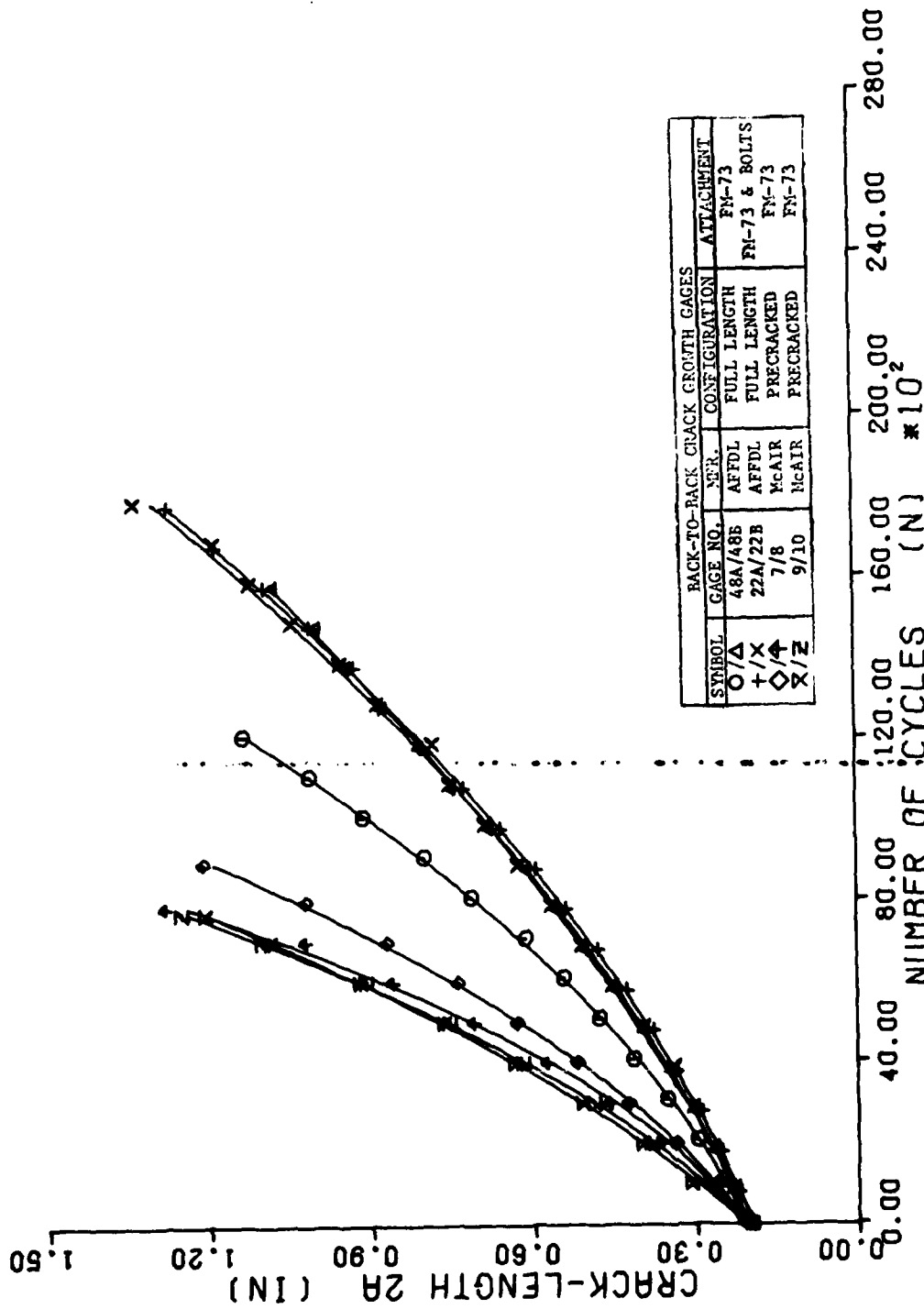


Figure 17. Constant Amplitude Tests - AFFDL Tests of AFFDL Full Length Gages (Precracking Tabs Attached) Compared with McAIR Tests of McAIR Gages

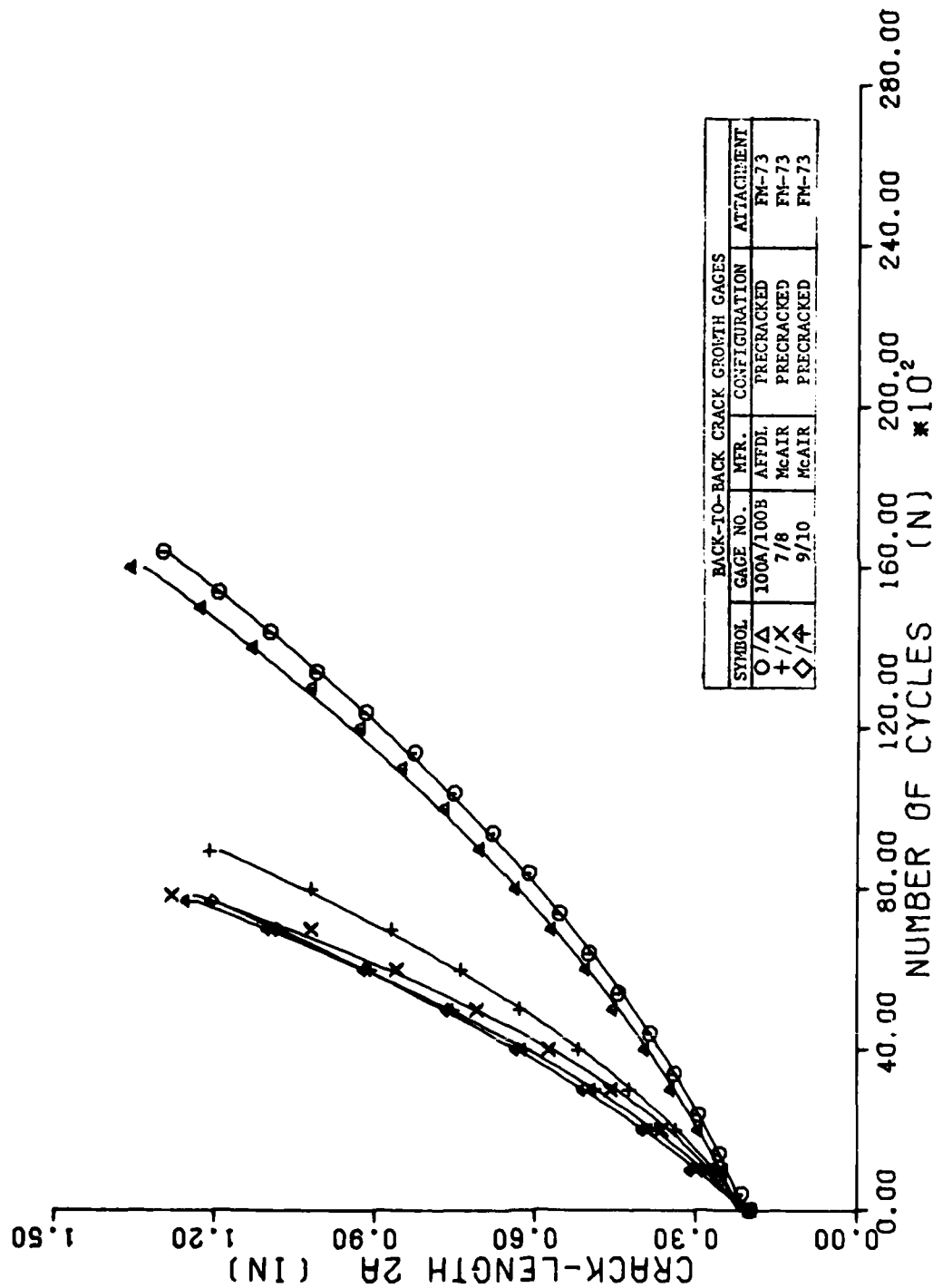


Figure 18. Constant Amplitude Tests - AFFDL Tests of AFFDL Gages Having Tighter Manufacturing Specifications Compared with McAir Tests of McAir Gages

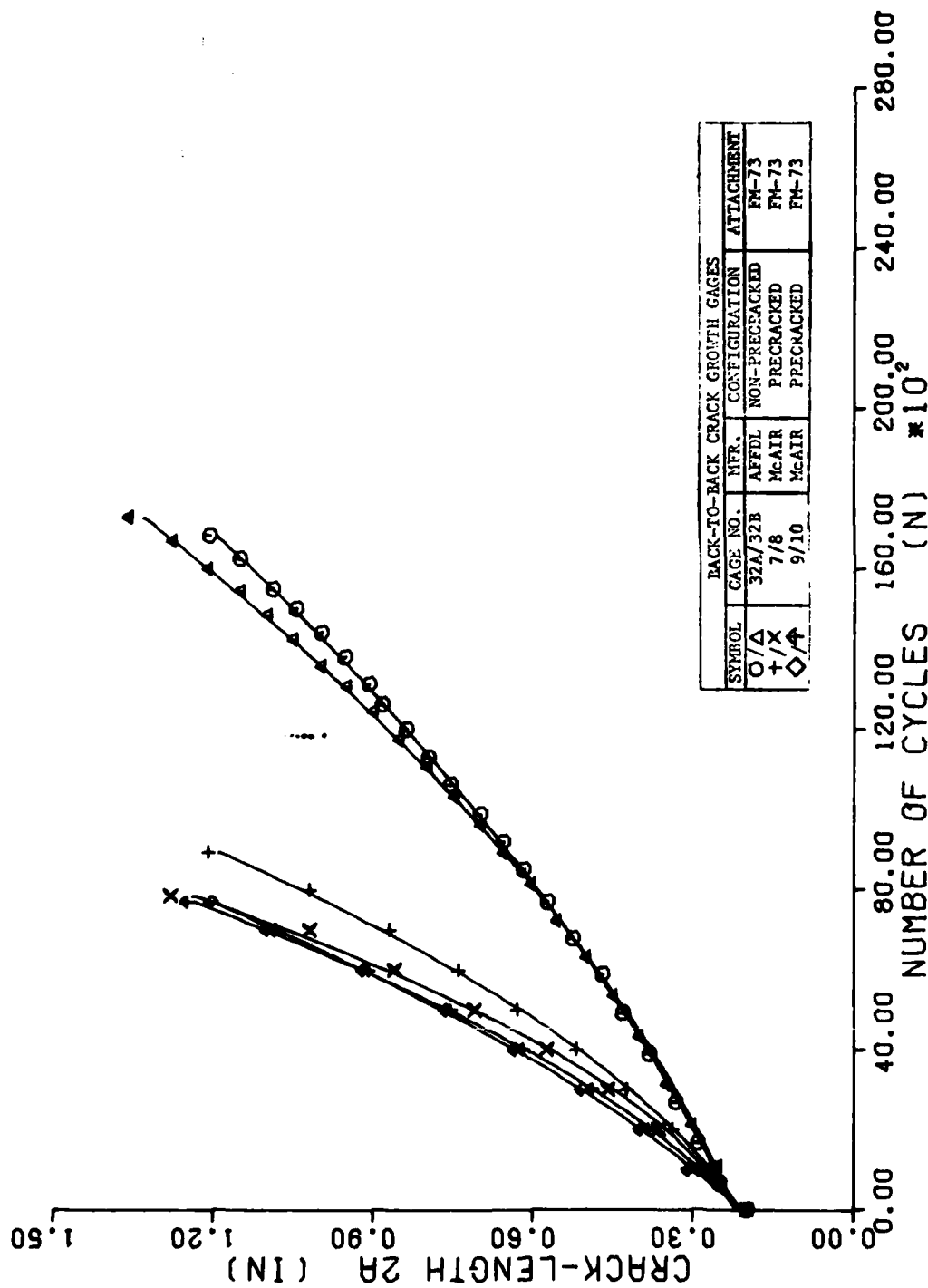


Figure 19. Constant Amplitude Tests - McAIR Tests of AFFDL Gages Compared with McAIR Tests of McAIR Gages

TABLE 1.

GAGE NO.	MANUFACTURER	TEST MATRIX CONFIGURATION	ATTACHMENT	TESTER
7	McAirt	Precracked	FM-73	McAirt
8	McAirt	Precracked	FM-73	McAirt
9	McAirt	Precracked	FM-73	McAirt
10	McAirt	Precracked	FM-73	McAirt
8A/B	AFFDL	Precracked	FM-73	AFFDL
30A/B	AFFDL	Precracked	FM-73	AFFDL
24A/B	AFFDL	Precracked	FM-73	AFFDL
18A/B	AFFDL	Non-Precracked	FM-73	AFFDL
M2A/B	McAirt	Precracked	FM-73	AFFDL
M4A/B	McAirt	Non-Precracked	FM-73	AFFDL
48A/B	AFFDL	Full-Length	FM-73	AFFDL
22A/B	AFFDL	Full-Length	FM-73 & Bolts	AFFDL
100A/B	AFFDL	Non-Precracked	FM-73	AFFDL
32A/B	AFFDL	Non-Precracked	FM-73	McAirt
118A/B	AFFDL	Non-Precracked	EA9309.1	AFFDL
110A/B	AFFDL	Full-Length	EA9309.1	AFFDL

NOTE: 1) All gages tested at 21.0 KSI

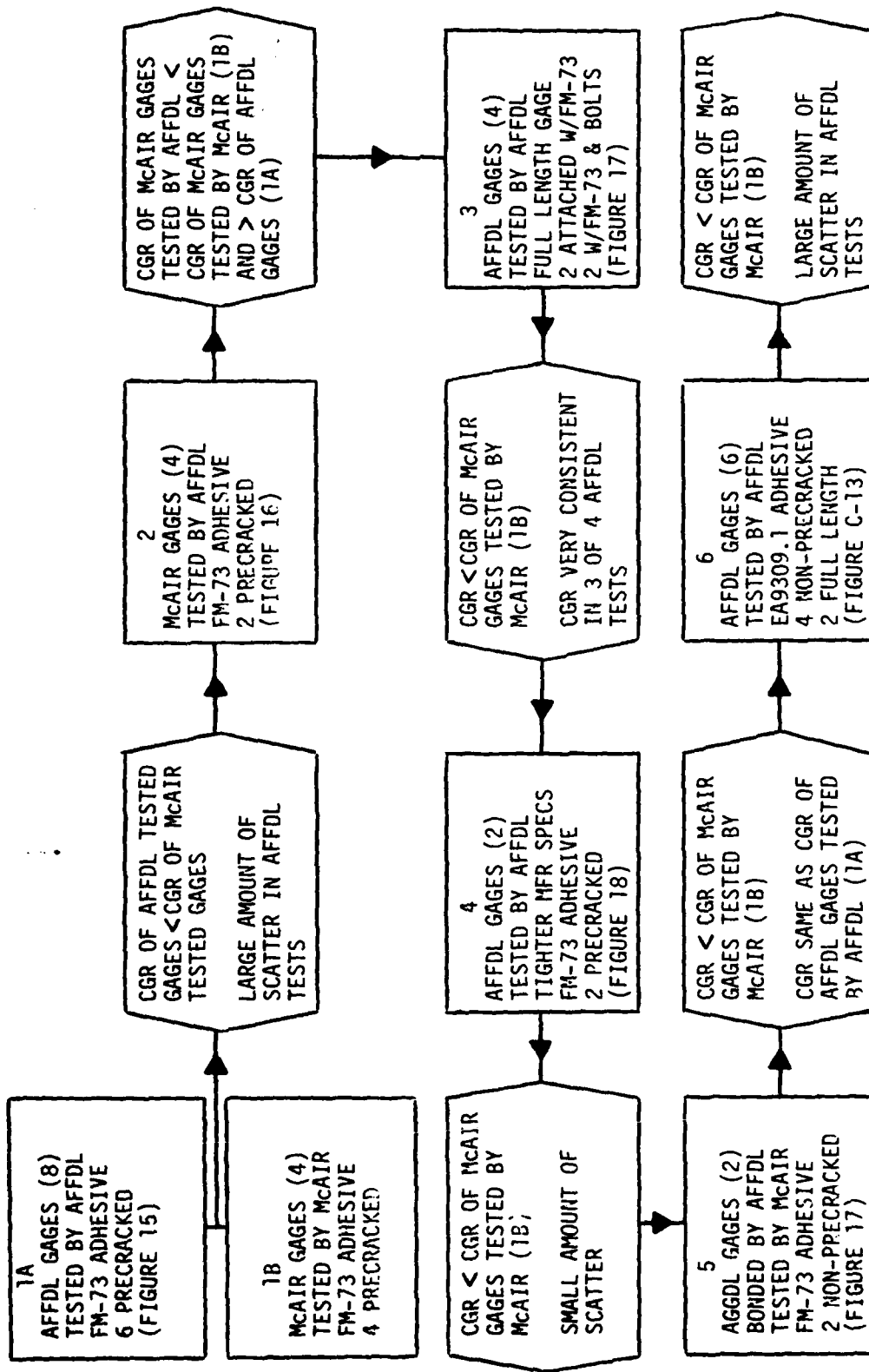
2) A/B indicates back-to-back gages on individual specimens (i.e., 8A and 8B).

3) Full length gages have precracking tabs attached (i.e., gage length is 7.05 inches).

4) All other gages 3.05 inches in length.

TABLE 2.

SUMMARY OF CONSTANT AMPLITUDE TEST PROGRAM



NOTE: CGR = Crack Growth Rate

2. Fatigue Test Article Results

At the equivalent of 4000 flight hours of baseline usage the F-4 full scale fatigue article test was put into a hold status in order to implement certain modifications in an attempt to extend the life of the aircraft. During this down time, McAir personnel installed four crack growth gages on the lower wing skin of the aircraft as explained in Sections 4, 4.1, and 4.2. No problems were encountered during the three day effort; however, soon after the McAir team departed, it was discovered that the FM-73 adhesive had not cured properly. Subsequent tests showed that the adhesive had not been heated to the recommended curing temperature of 250°F. The problem was the result of faulty thermocouple placement and insufficient heating blanket capacity. McAir later returned, stripped off the gages, and rebonded them using a larger heating blanket. Additional thermocouples were used to ensure the proper cure temperature was reached and also to ensure that the wing skin was not heated above 350°F in the vicinity of the gages or above 200°F in the vicinity of the fastener patterns.

During this time a team from Douglas Aircraft Company bonded a fifth crack growth gage on the lower skin of the left wing in a location duplicating the McAir site 1 gage on the right wing (See Figure 13). The fifth gage was bonded after using a surface preparation designed for environmental tests. The objective of the fifth gage was to demonstrate a non-tank phosphoric acid anodize surface preparation. A McAir crack growth gage was used for this test.

Since the crack growth gage was designed for a life of 12,000 hours with a measurable crack growth in one thousand hours, it was decided that Fax-Film measurements would be taken at 500 hour intervals. After the first 500 hours of equivalent flight time, only two of the five gages showed any significant

crack growth. Cycling of the test article continued with Fax-Film measurements being taken at 1000 and 1500 hours of simulated flight. During this thousand hour interval (i.e., from 500 to 1500 hours) no new crack growth was detected. A strain gage was attached to one of the crack growth gages to determine if the proper load transfer was occurring through the adhesive. Results of this strain survey showed no load going to the crack growth gage. A visual inspection was made and all five crack growth gages were found to be debonded.

Several theories were investigated as the possible cause for debonding. These included: (1) the surface of the wing skin was improperly prepared prior to bonding, (2) the adhesive was improperly cured during the bonding process (i.e., the proper curing temperature was not reached), and (3) the adhesive in its raw form had absorbed moisture prior to the bonding process and thus the quality of the bond was reduced. (Later analysis of the adhesive showed the presence of silicone contaminants in the adhesive. It was also later discovered that the entire bare metal-skinned fatigue article had been previously coated with a silicone spray as a corrosion prohibitor. This could be a possible explanation as to why debonding occurred.)

While the above theories were being investigated, it was decided that the crack growth gages would be rebonded to the fatigue article during the next down time for aircraft inspection. This inspection came after 6000 hours of baseline flight.

The heating blanket procedure used in the first bonding process could not be used again because of the numerous loading pads attached to the fatigue article. During the first bonding, the load pads were removed to facilitate

heating the required area to the curing temperature. EA9309.1, a two-part room temperature cure adhesive from the Hysol Division of the Dexter Corporation, was chosen for the rebonding process. Since this room temperature cure adhesive does not have the strength, temperature resistance or environmental durability of FM-73 (a heat cure adhesive), more constant amplitude tests were completed to determine adhesive performance under cyclic loading (see Appendix C). At this time in the program, it was decided to drop the MIL-STD-810C environmental tests and to concentrate on an attempt to obtain usable data from the gages bonded to the fatigue article.

Eight crack growth gages were attached to the lower wing skin of the F-4 fatigue article using EA9309.1 adhesive. Bonding was performed by AFFDL personnel. Five AFFDL machined gages and three McAir chem-milled gages were used. Five gages were bonded to the lower wing skin of the right wing; three gages were attached to the left wing. Gage locations are shown in Figure 20. Gages bonded to the left wing were in duplicate locations of sites on the right wing. Five crack growth gages were strain gaged to determine if load was being transferred through the adhesive.

Raw crack growth data from the crack growth gages attached to the fatigue article with EA9309.1 adhesive is shown in Table 3. Plots of actual crack growth compared with McAir predictions are shown in Figures 21 through 25. In general, the experimental measurements agree quite well with McAir predictions.

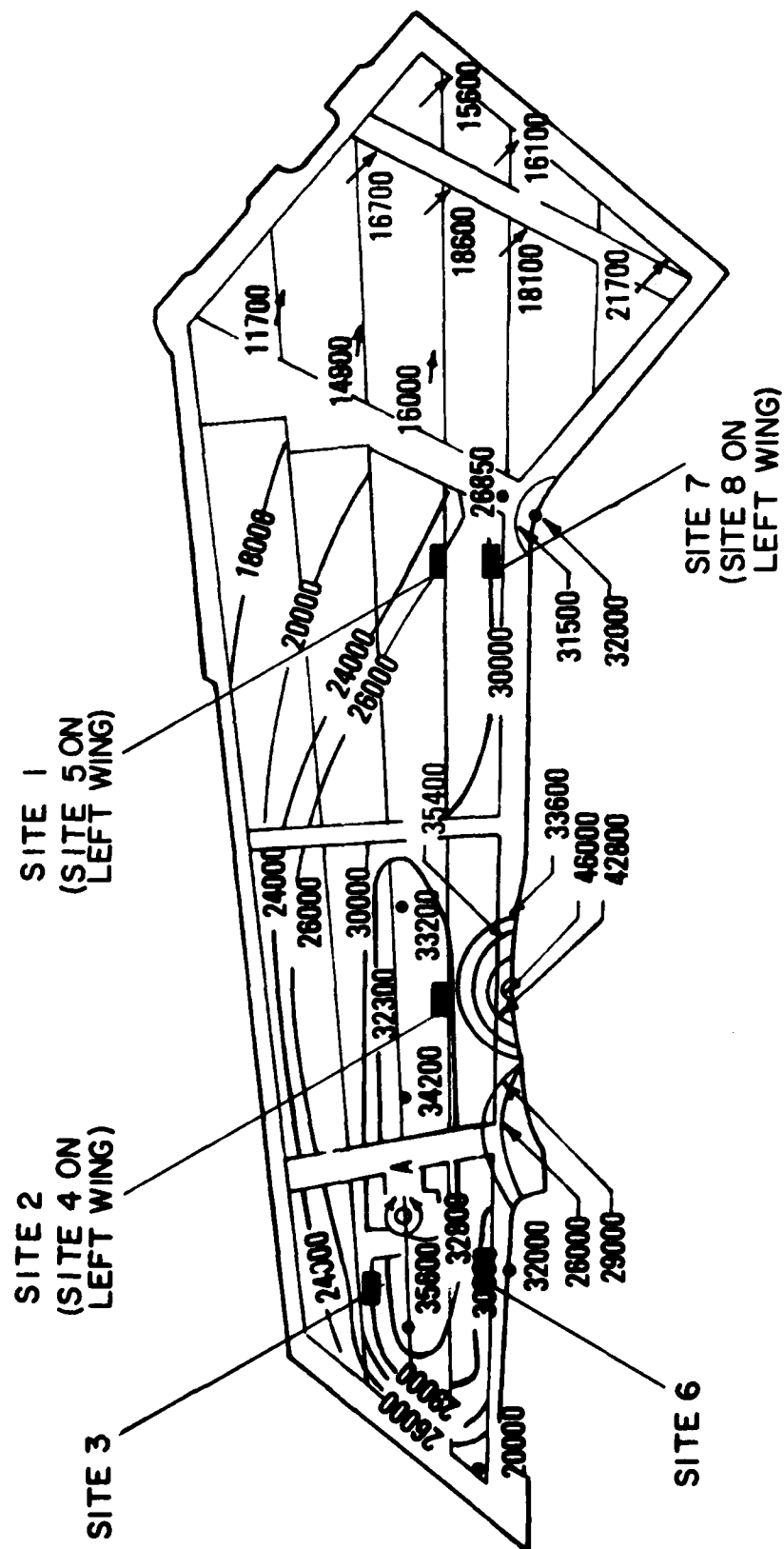


Figure 20. Selected Sites on the Lower Wing Skin of the F-4 Fatigue Article for Bonding Crack Growth Gages with EA9309.1 Adhesive

TABLE 3. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES BONDED TO F-4 FATIGUE TEST ARTICLE WITH 59309.1 ADHESIVE

SITE	1	2	3	4	5	6	7	8
GAGE	52	53	54	104B	102A	148	146B	102B
MFR	McAirt	McAirt	McAirt	AFFDL	AFFDL	AFFDL	AFFDL	AFFDL
STRESS (KSI)	26	33	30	33	26	30	30	30
FLT HRS	2a (INCHES)							
BASLINE	0.197	0.197	0.194	0.190	0.427	0.379	0.223	0.246
438	0.200	0.217	0.195	BAD FAX-FILM	0.446	BAD FAX-FILM	0.259	0.270
1000	0.201	0.223	0.198	0.222	0.465	0.415	0.227	0.296
1525	0.204	0.246	VISUAL DEBOND	0.229	0.481	0.442	0.280	0.303
2000	0.207	0.252		VISUAL DEBOND	0.499	0.424	VISUAL DEBOND	0.315
2500	NO DATA	NO DATA			NO DATA	NO DATA		NO DATA
3000	NO DATA	NO DATA			NO DATA	NO DATA		NO DATA
3500	VISUAL DEBOND	BAD FAX-FILM			0.533	VISUAL DEBOND		0.368
4000		0.286			0.544			0.383
4520		NO CHANGE			0.550			0.385
5000					0.564			NO CHANGE
5566					0.575			↓
6000					0.590			0.392

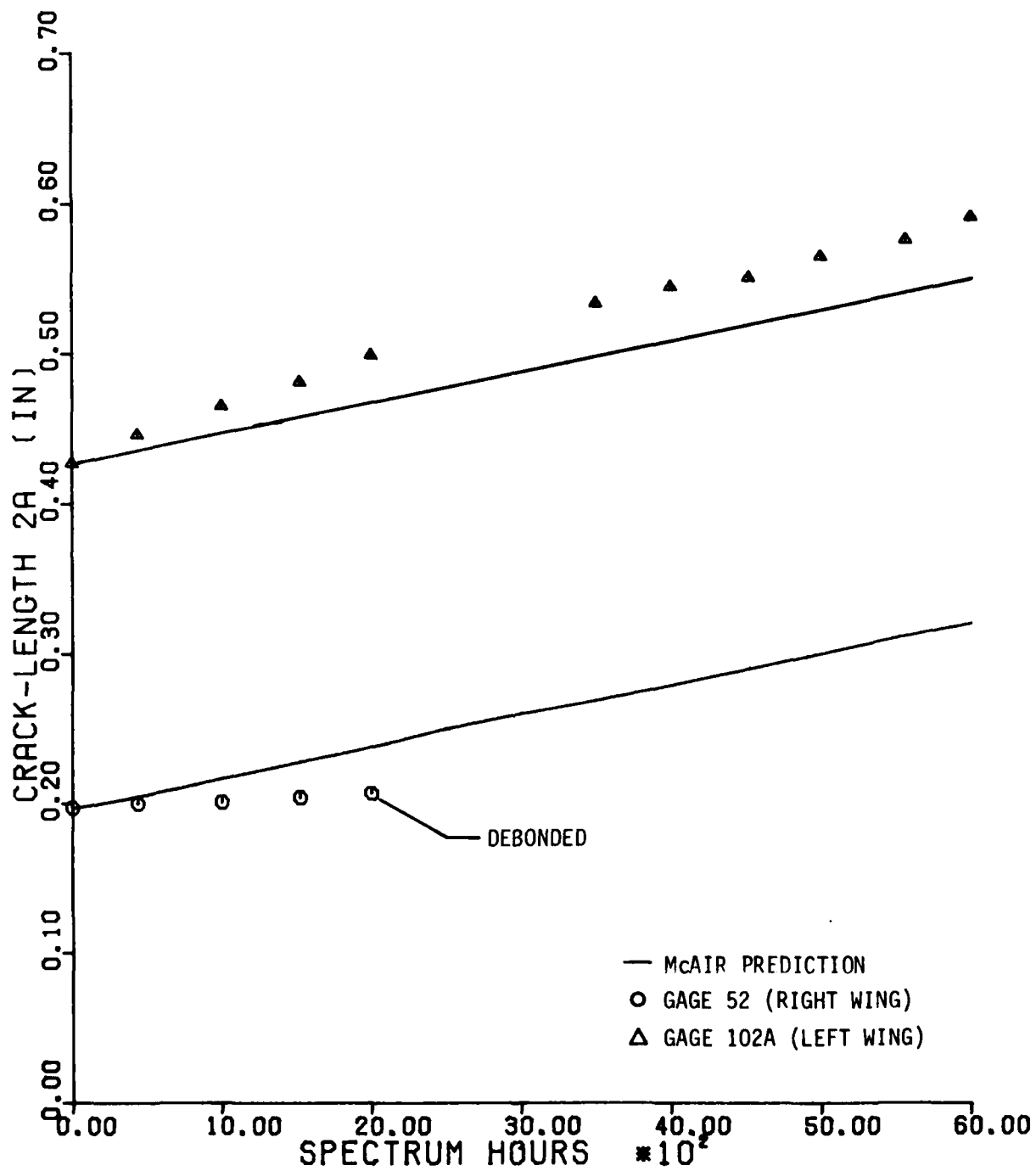


Figure 21. Comparison of Crack Length Measurements and Predictions for Site 1 (Site 5 on Left Wing)

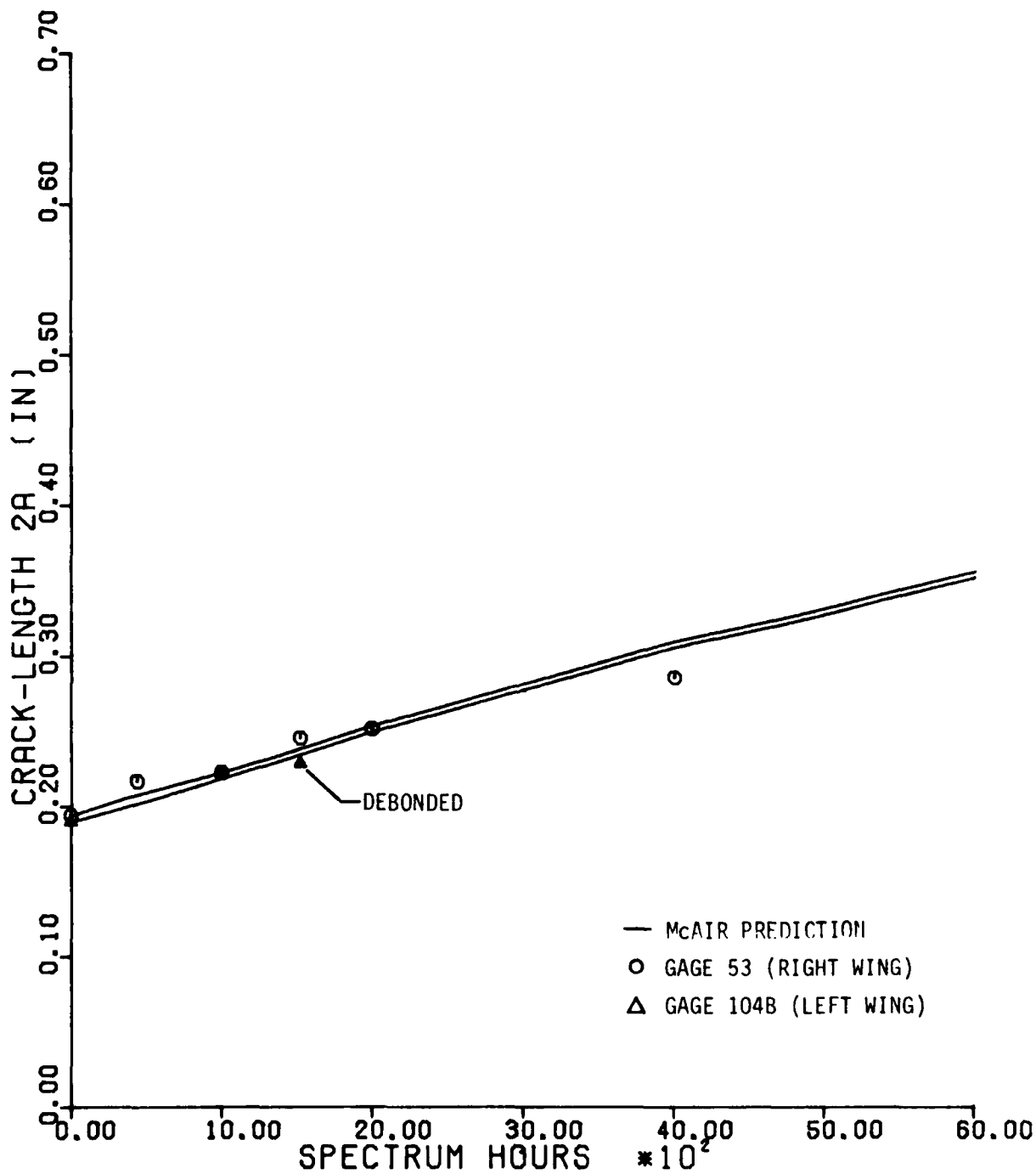


Figure 22. Comparison of Crack Length Measurements and Predictions for Site 2 (Site 4 on Left Wing)

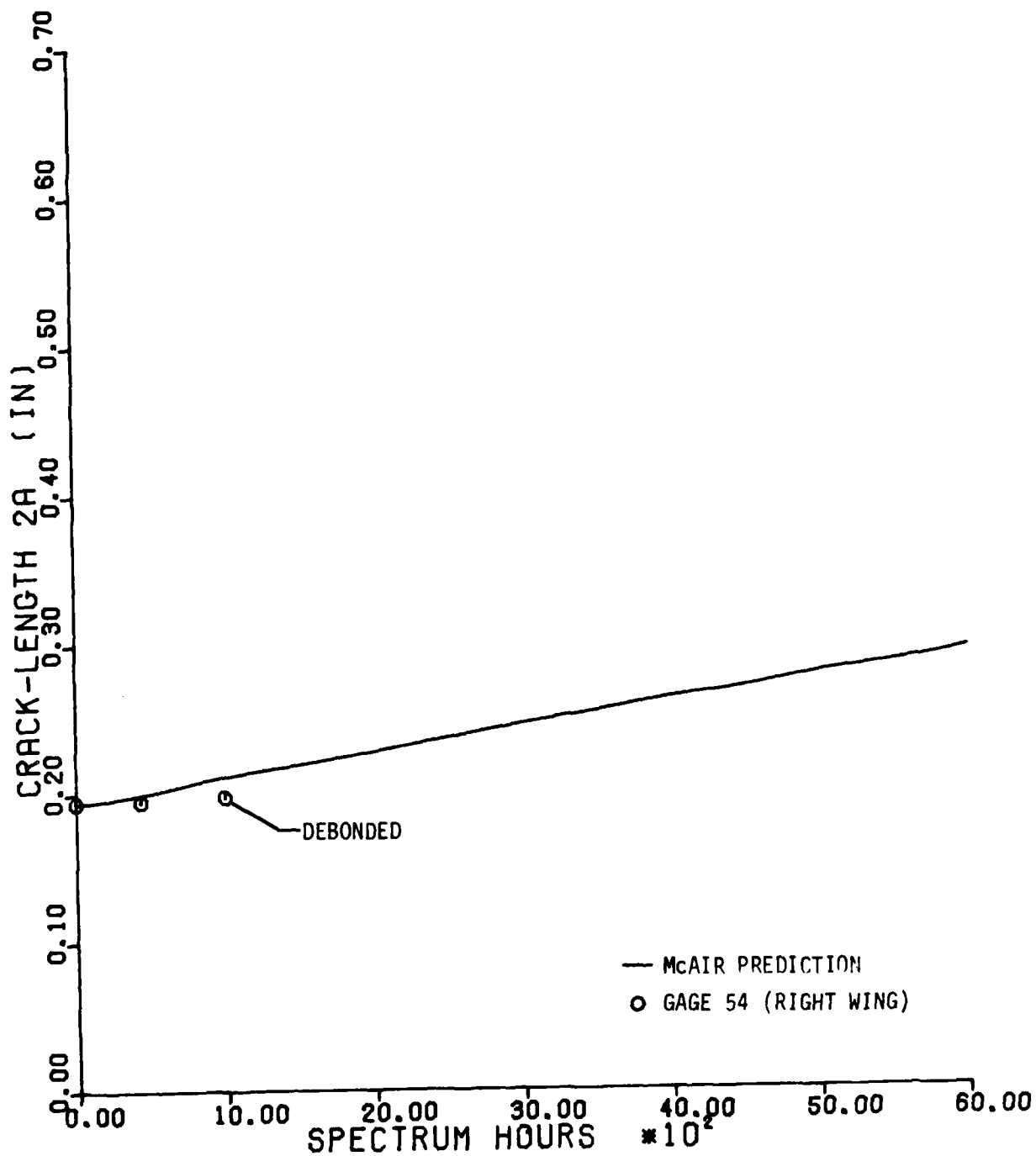


Figure 23. Comparison of Crack Length Measurements and Predictions for Site 3

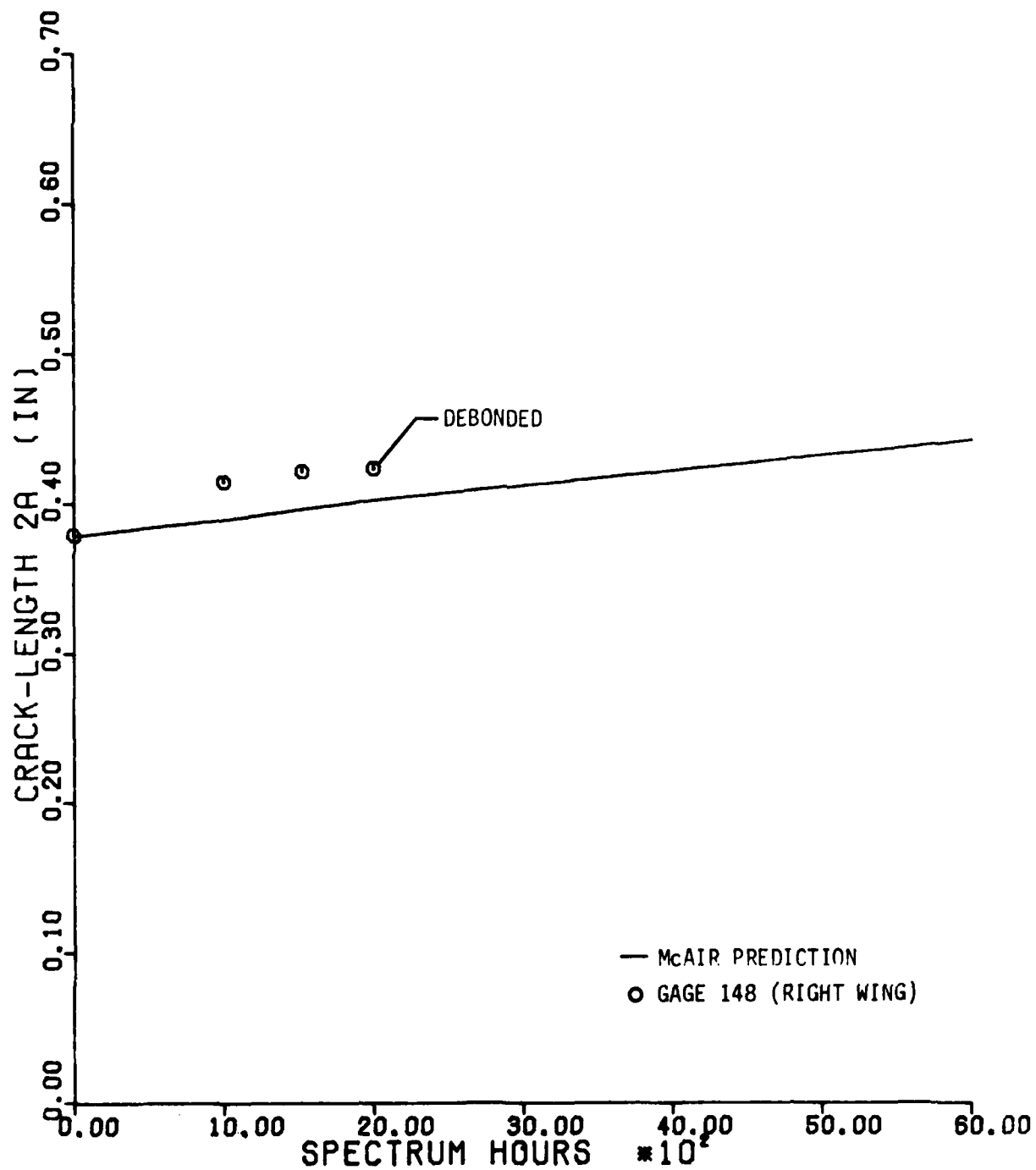


Figure 24. Comparison of Crack Length Measurements and Predictions for Site 6

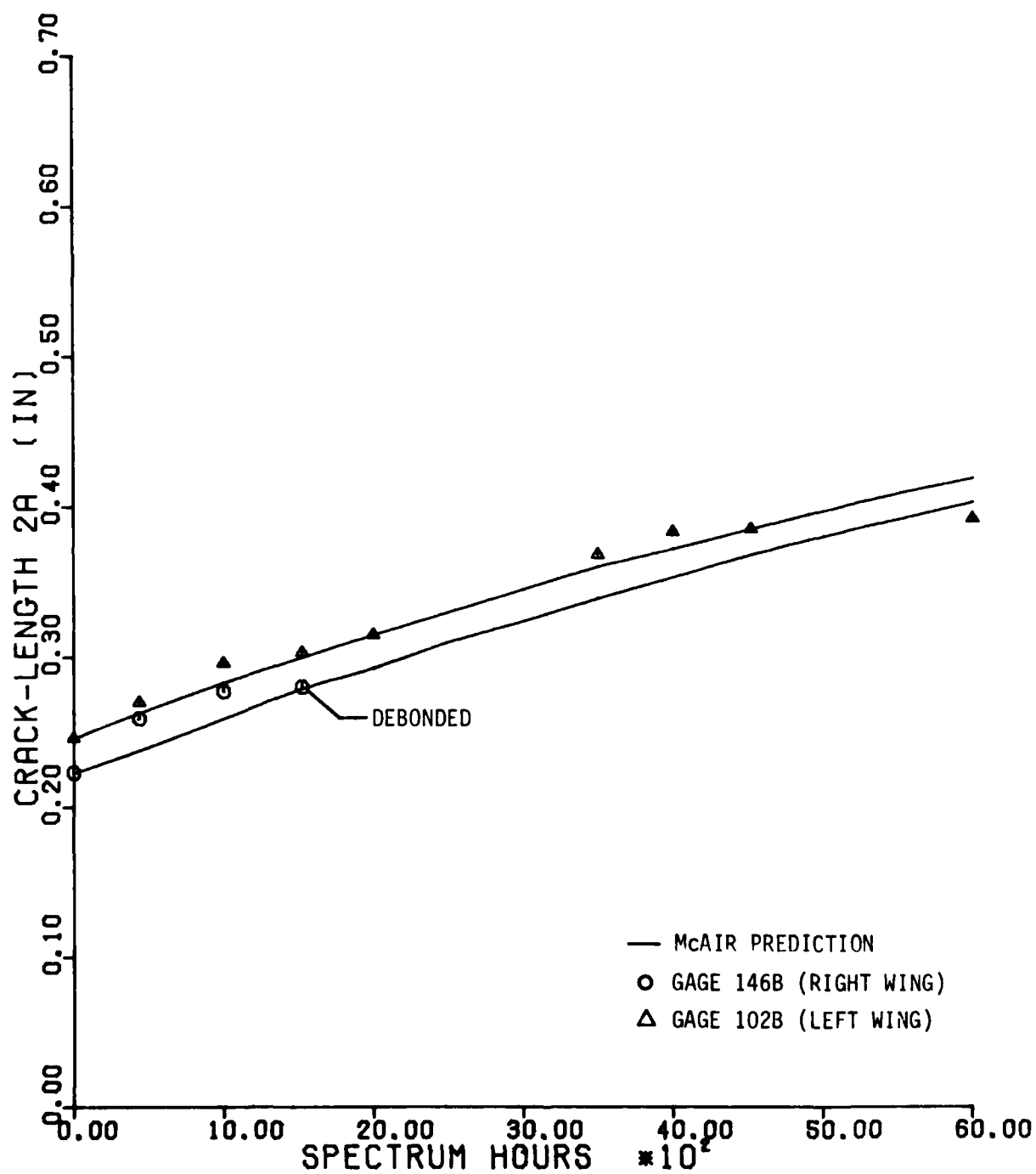


Figure 25. Comparison of Crack Length Measurements and Predictions for Site 7 (Site 8 on Left Wing)

SECTION IV

CONCLUSIONS

Results obtained from this program suggest that crack growth in crack growth gages adhesively bonded to an aircraft is generally predictable using techniques developed by McAir personnel. However, there are problems that still exist in the crack growth gage concept. Considerable further research is required to (1) develop a simple procedure for reliably bonding gages to an aircraft structure, (2) demonstrate reproducibility of crack growth from gage to gage, and (3) complete gage qualification and service evaluation testing. These tasks must be completed before the crack growth gage can be implemented fleet-wide to track aircraft service life.

The main observations of this program are summarized in the following paragraphs.

1. Bonding of the crack growth gages to the F-4 full-scale fatigue article with FM-73, a heat cure adhesive, was unsuccessful due to circumstances peculiar to this structure. Investigation led to the discovery that the entire aircraft structure had been coated with a silicone spray. This is a possible explanation as to why debonding occurred within a very short period of time (i.e., less than 1500 flight hours). Therefore, this test did not aid in qualifying FM-73 adhesive for bonding of the gages to fleet aircraft.

2. Although it was known that EA-9309.1, a room temperature cure adhesive, did not have the strength, temperature resistance, or environmental durability of FM-73, EA9309.1 was used to bond eight crack growth gages to the fatigue article in an attempt to obtain usable data from the full-scale

test. Because of durability limitations of EA9309.1, three of the eight gages had visually debonded after 2000 hours of simulated flight. After 4000 spectrum hours only three gages (one McAir gage; two AFFDL gages) remained attached to the structure. Data obtained from these remaining gages generally agreed with behavior predicted by McAir personnel.

3. Considerable variations were noted between constant amplitude tests performed by McAir and AFFDL personnel. Crack growth rates were considerably slower for gages manufactured and bonded by AFFDL personnel. A large amount of scatter was observed in the AFFDL constant amplitude tests. Unfortunately, no definite explanations could be found for these dissimilarities.

4. The Fax-Film method of recording crack length in the gages was found to provide an adequate replica of the crack.

SECTION V

RECOMMENDATIONS

Due to inconsistencies in constant amplitude test results and problems encountered during bonding of the gages to the F-4 full-scale fatigue article, it is recommended that research and development of the crack growth gage as a possible fleet-wide tracking device be continued.

It is also recommended that further research be undertaken to:

1. Develop a simple procedure for reliably bonding crack growth gages to an aircraft structure.
2. Demonstrate reproducibility of crack growth from gage to gage.
3. Complete a comprehensive gage qualification test program in accordance with MIL-STD-810C requirements.
4. Complete a comprehensive gage qualification test program defining the operational parameters and limitations for using Fax-Film under actual field conditions.
5. Determine the protection required (i.e., cover, sealant, paint, etc.) for an externally mounted gage.
6. Consider developing a crack growth gage with a life less than the life of the aircraft (eg, a gage that would last 1000 or 2000 hours). This type of gage could have the capability of producing more data points throughout the life of the aircraft than a gage that was developed to last the design life of the aircraft.

APPENDIX A

da/dN COUPON TEST RESULTS

This appendix contains tabulated data (Tables A-1 and A-2) and log-log plots of da/dN versus ΔK (Figure A-1) for the da/dN coupon tests.

TABLE A-1. da/dN COUPON TESTS

da/dN Specimen	1	2	3	4
TEST SECTION THICKNESS(IN)	0.041	0.041	0.038	0.040
SLOT LENGTH (IN)	0.098	0.093	0.104	0.104
P _{max} (LBS)	1000	900	700	700
P _{min} (LBS)	0	0	0	0
CYCLIC RATE (Hz)	2.5	2.5	2.5	2.5

TABLE A-2. CRACK PROPAGATION DATA FOR da/dN COUPON TESTS

da/dN - 1		da/dN - 2		da/dN - 3		da/dN - 4	
N	2a (in.)	N	2a(in.)	N	2a(in.)	N	2a(in.)
0	0.098	0	0.093	0	.104	0	0.104
9000	0.135	11000	0.100	25000	.108	50000	0.197
14000	0.219	12000	0.110	29000	.122	52000	0.210
16000	0.274	13000	0.121	30000	.132	54000	0.221
18000	0.350	14000	0.128	31000	.138	56000	0.243
20000	0.454	15000	0.131	32000	.144	57000	0.249
22000	0.625	16000	0.133	34000	.154	58000	0.261
24000	0.914	17000	0.139	36000	.169	59000	0.271
		18000	0.145	36000	.197	60000	0.282
		19000	0.157	40000	.212	61000	0.295
		20000	0.169	42000	.239	62000	0.306
		21000	0.184	44000	.272	63000	0.316
		22000	0.194	46000	.306	64000	0.336
		23000	0.211	48000	.345	65000	0.345
		24000	0.224	50000	.390	66000	0.372
		25000	0.244	52000	.437	67000	0.393
		26000	0.257	54000	.513	68000	0.413
		27000	0.272	55000	0.551	69000	0.446
		28000	0.299	56000	0.600	70000	0.472
		29000	0.327	57000	0.653	71000	0.500
		30000	0.357	58000	0.712	72000	0.535
		31000	0.385	59000	0.789	73000	0.568
		32000	0.427	60000	0.884	74000	0.609
		33000	0.465	61000	1.017	75000	0.661
		34000	0.519	61500	1.164	76000	0.714
		35000	0.570			77000	0.767
		36000	0.630			78000	0.828
		37000	0.705			79000	0.920
		37500	0.760			80000	1.005
		38000	0.857				
		38500	0.930				

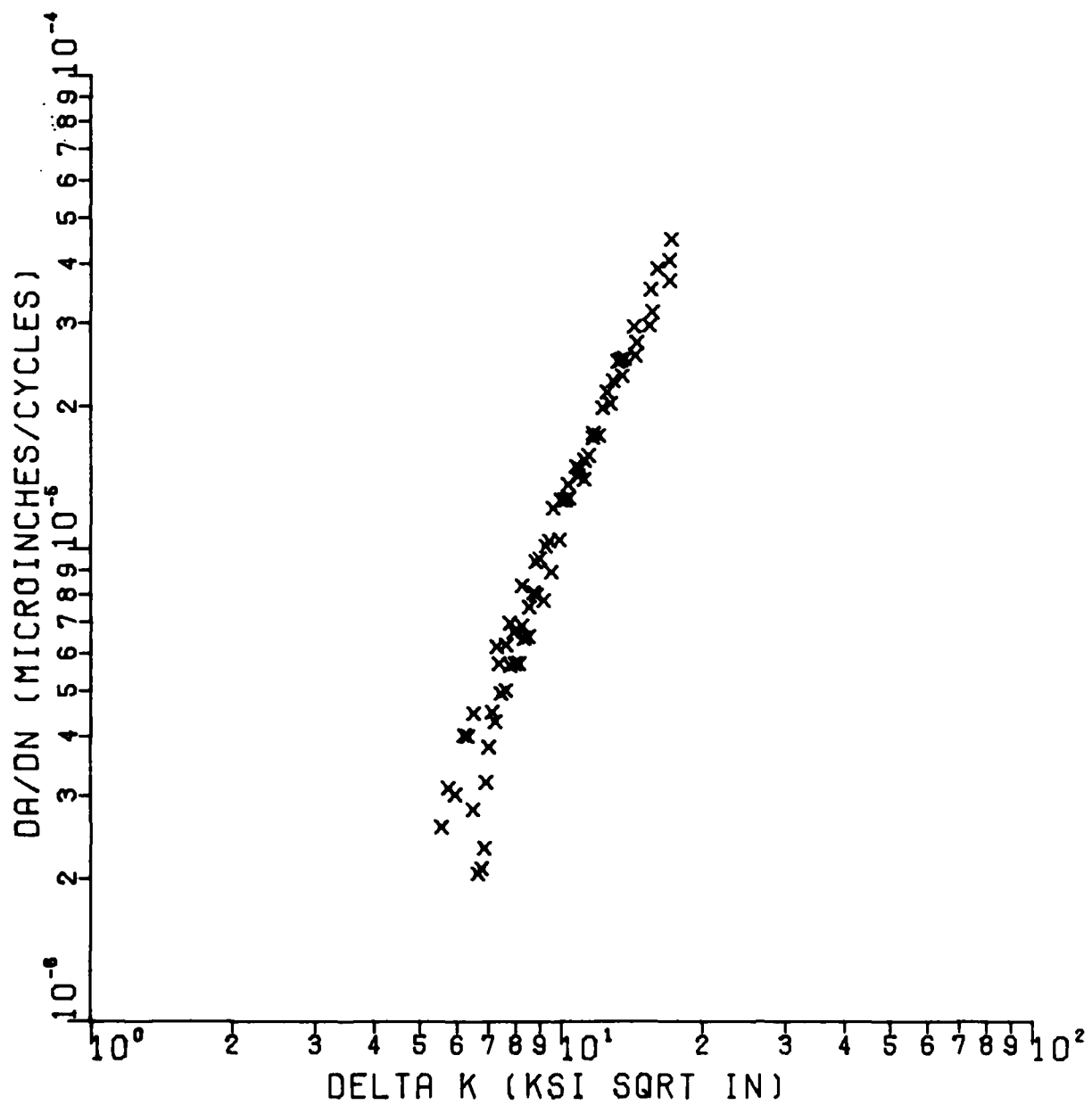


Figure A-1. Crack Growth Rate for 7075-T6 da/dN Coupons

APPENDIX B

STRAIN MEASUREMENTS

Prior to testing of the slotted crack growth gages, one carrier specimen and its adhesively bonded crack growth gages were extensively instrumented with strain gages to measure stresses on the specimen as well as the stresses transferred from the specimen through the adhesive to the crack growth gage. Strain gage locations are shown in Figure B-1. Strain measurements taken under static load conditions are shown in Table B-1.

During the primary testing of the slotted crack growth gages, several carrier specimens and crack growth gages were instrumented with strain gages. Strain gage locations are shown in Figure B-2. Strain measurements taken under static load conditions are shown in Table B-2.

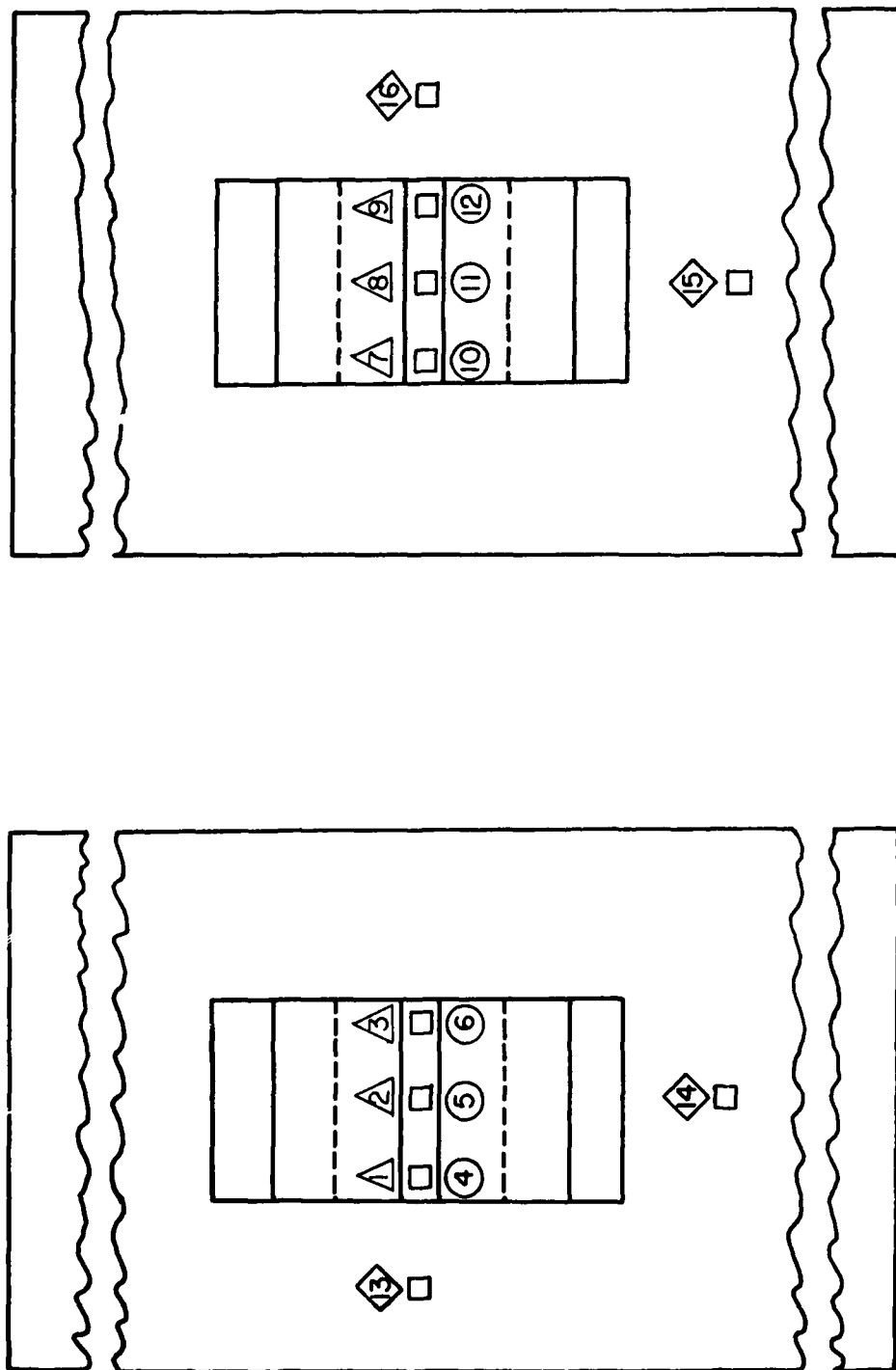


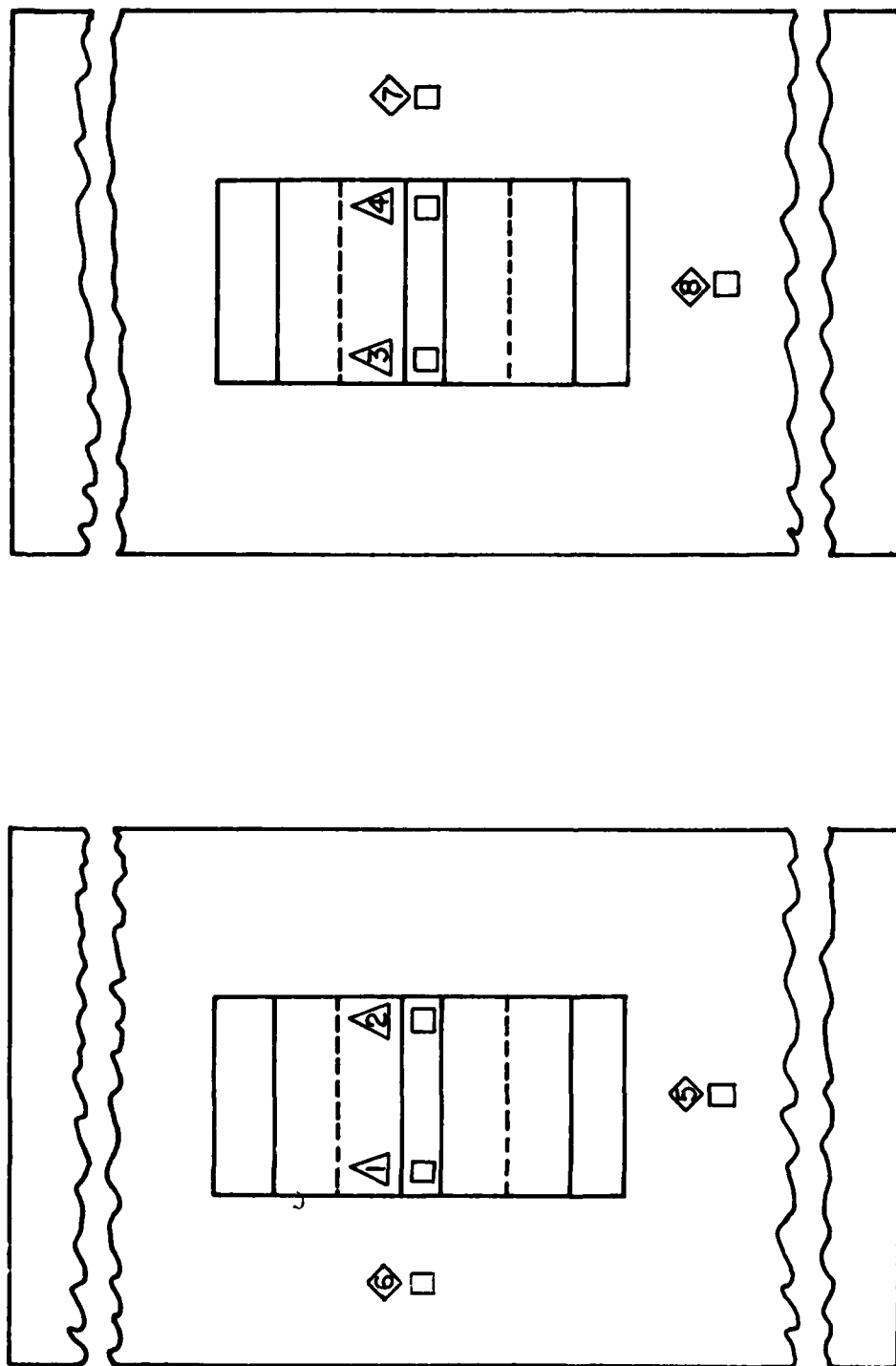
Figure B-1. Strain Gage Locations on Load Transfer Specimen

TABLE B-1. STRAIN SURVEY RESULTS FOR LOAD TRANSFER SPECIMEN 1-1-1

SPECIMEN CONDITION	STRESS (KSI)	LOAD (KIPS)	STRAIN (MICRO-INCHES)															
			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Prior to Fatigue Cycling	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	5	6695	650	620	605	689	660	665	750	725	735	695	720	735	460	500	515	480
	10	13390	1310	1250	1230	1380	1330	1340	1500	1465	1480	1400	1455	1475	925	1015	1035	965
	15	20085	2070	1970	1940	2165	2085	2095	2345	2285	2305	2195	2275	2310	1450	1590	1620	1505
	20	26780	2575	2550	2515	2790	2690	2710	3020	2950	2975	2840	2930	2975	1875	2050	2085	1940
After 22000 Fatigue Cycles	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	5	6695	655	625	615	700	670	675	755	755	735	705	730	750	470	505	525	485
	10	13390	1325	1275	1250	1410	1350	1360	1520	1495	1485	1425	1480	1500	950	1040	1065	985
	15	20085	2095	2000	1980	2205	2120	2135	2370	2335	2335	2230	2310	2340	1490	1630	1655	1535
	20	26780	2680	2565	2535	2815	2710	2730	3035	2985	2990	2760	2965	3005	1920	2100	2120	1970
After 42000 Fatigue Cycles	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	5	6695	630	600	595	670	650	650	730	715	715	680	715	730	470	510	515	475
	10	13390	1330	1275	1255	1400	1355	1355	1515	1480	1490	1420	1480	1500	965	1050	1075	975
	15	20085	2050	1965	1940	2155	2080	2085	2325	2275	2290	2185	2275	2300	1480	1615	1615	1505
	20	26780	2595	2480	2450	2720	2625	2630	2925	2865	2885	2755	2855	2890	1865	2020	2090	1920
After 62000 Fatigue Cycles	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	5	6695	650	620	615	685	665	670	750	720	735	705	730	710	440	500	515	475
	10	13390	1310	1265	1245	1390	1340	1450	1510	1460	1475	1410	1450	1450	910	1015	1035	975
	15	20085	2055	1980	1950	2165	1985	2205	2345	2275	2310	2205	2265	2275	1440	1590	1610	1505
	20	26780	2690	2580	2545	2820	2620	2840	3040	2965	3000	2865	2960	3060	1910	2070	2090	1920
After 84000 Fatigue Cycles	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	5	6695	665	630	625	700	680	685	760	740	745	710	735	740	500	530	530	480
	10	13390	1325	1265	1245	1395	1350	1355	1510	1475	1485	1415	1470	1480	1025	1045	1055	965
	15	20085	1990	1905	1885	2095	2025	2035	2265	2210	2230	2130	2205	2220	1415	1570	1575	1505
	20	26780	2665	2550	2525	2790	2695	2710	3005	2940	2965	2835	2930	2950	2015	2090	2090	1940
After 102000 Fatigue Cycles	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	5	6695	650	625	620	700	675	680	770	755	715	680	745	745	515	525	530	480
	10	13390	1295	1245	1235	1390	1340	1350	1525	1540	1415	1350	1505	1505	990	1045	1050	965
	15	20085	1935	1860	1840	2070	1990	1905	2260	2300	2110	2110	2245	2245	1455	1555	1560	1505
	20	26780	2605	2500	4215	2770	2670	2585	3015	3090	2825	2825	3020	3020	2100	2085	2090	1940

△ Wire Broke Lose

△ Strain Gage Failed



SIDE A
 SIDE B
 Δ - STRAIN GAGES LOCATED ON FRONT OF CRACK GAGE
 ◇ - STRAIN GAGES LOCATED ON CARRIER SPECIMEN

Figure B-2. Strain Gage Locations on Slotted Crack Growth Gages

TABLE B-2. STRAIN SURVEY RESULTS FOR THREE CRACK GROWTH GAGES PRIOR TO CYCLIC TESTING

SPECIMEN NUMBER	STRESS (KSI)	LOAD (KIPS)	STRAIN (MICRO-INCHES)							
			1	2	3	4	5	6	7	8
100A/100B	0	0	0	0	0	0	0	0	0	0
	5	6695	645	630	695	720	510	445	470	545
	10	13390	1290	1275	1375	1430	1030	900	940	1085
	15	20085	2005	1980	2125	2210	1600	1400	1460	1680
	21	28120	2710	2680	2855	2965	2170	1900	1975	2270
110A/110B	0	0	0	0	0	0	0	0	0	0
	5	6695	600	599	660	639	519	395	412	510
	10	13390	1225	1220	1303	1269	1016	812	838	1007
	15	20085	1920	1907	2022	1973	1574	1280	1304	1565
	21	28120	2605	2584	2722	2660	2131	1742	1772	2106
48A/48B	0	0	0	0	0	0	0	0	0	0
	5	6695	635	615	710	745	560	440	470	535
	10	13390	1275	1335	1425	1510	1130	890	945	1065
	15	20085	1975	2010	2190	2235	1695	1345	1445	1620
	21	28120	2660	2770	2920	3110	2310	1895	1950	2205

APPENDIX C

CONSTANT AMPLITUDE TEST RESULTS

This appendix contains tabulated results and graphical plots of crack length versus cycles for each specimen tested. The crack propagation results are presented in Tables C-1 through C-12. Plots of the test data are presented in Figures C-1 through C-12. Figure C-13 shows the results of constant amplitude tests performed on crack growth gages bonded with EA9309.1 adhesive compared with the McAir test results on gages bonded with FM-73 adhesive.

TABLE C-1. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 8A AND 8B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 HZ	
CRACK GROWTH GAGE 8A		CRACK GROWTH GAGE 8B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.105	EDM	0.103
PRECRACK	0.151	PRECRACK	0.113
1000	0.190	1000	0.136
2000	0.234	2000	0.158
3000	0.277	3000	0.183
4000	0.319	4000	0.207
5000	0.366	5000	0.234
6000	0.417	6000	0.263
7000	0.471	7000	0.294
8000	0.525	8000	0.322
9000	0.590	9000	0.359
10000	0.671	10000	0.402
11000	0.744	11000	0.439
12000	0.798	12000	0.479
13000	0.917	13000	0.545
14000	1.009	14000	0.574
15000	1.126	15000	0.611
16000		16000	0.640
17000		17000	0.667
18000		18000	0.731
19000		19000	0.756
20000		20000	0.787
21000		21000	0.835
22000		22000	0.874
23000		23000	0.926
24000		24000	0.969
25000		25000	1.036
26000		26000	1.084

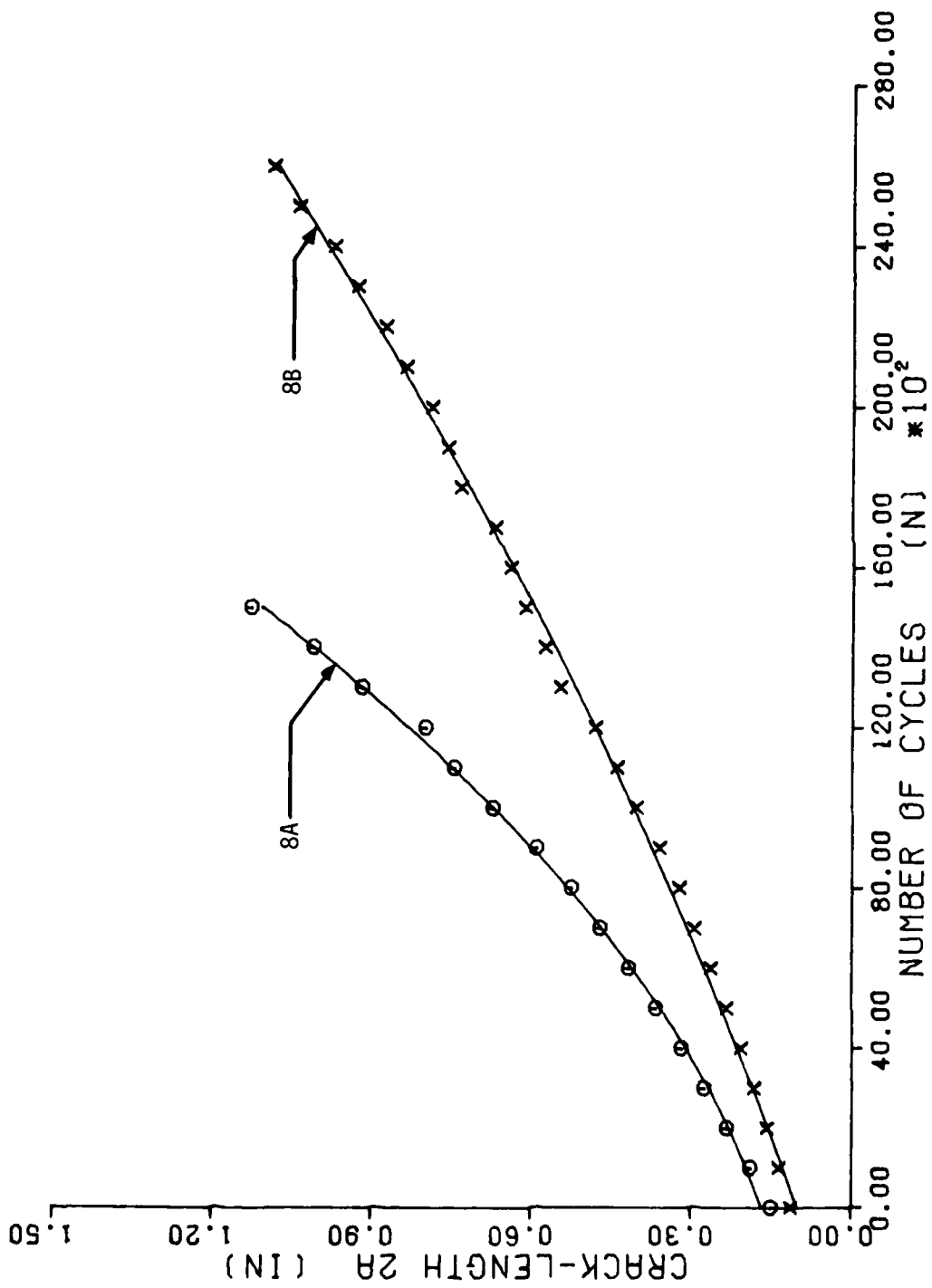


Figure C-1. Constant Amplitude Tests of Crack Growth Gages 8A and 8B

TABLE C-2 CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 18A AND 18B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 1.0 Hz	
CRACK GROWTH GAGE 18A		CRACK GROWTH GAGE 18B	
<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>	<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>
EDM	0.100	EDM	0.103
2000	0.117	2000	0.117
4000	0.155	4000	0.164
6000	0.208	6000	0.216
8000	0.287	8000	0.294
10000	0.353	10000	0.378
11000	0.391	11000	0.414
12000	0.436	12000	0.467
13000	0.487	13000	0.527
14000	0.540	14000	0.590
15000	0.592	15000	0.647
16000	0.645	16000	0.711
17000	0.716	17000	0.755
18000	0.773	18000	0.858
19000	0.826	19000	0.931
20000	0.888	20000	1.003
21000	0.945	21000	1.096
22000	1.010	22000	1.185
23000	1.074	23000	1.297
24000	1.134		
25000	1.208		
26000	1.293		

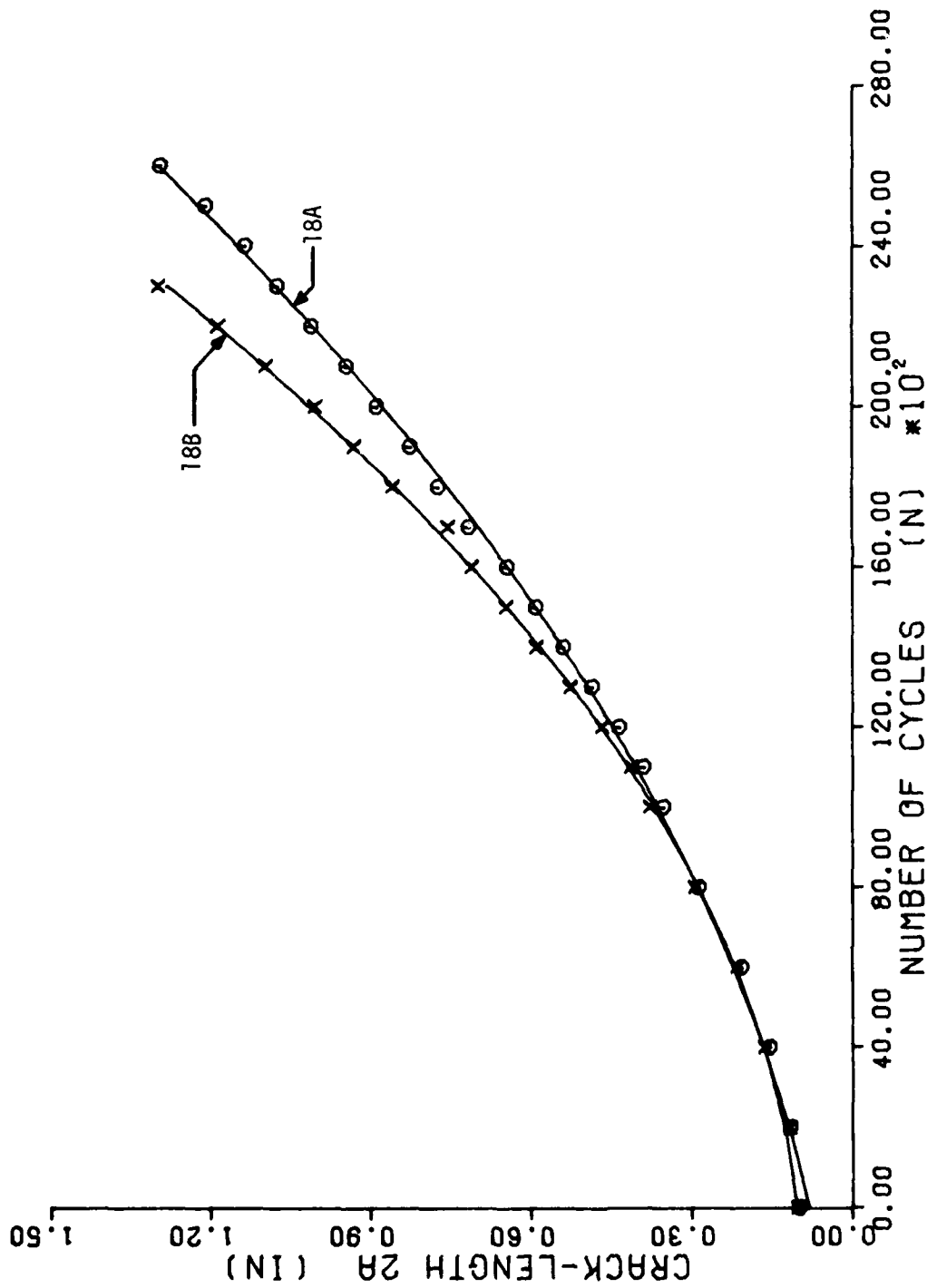


Figure C-2. Constant Amplitude Test of Crack Growth Gages 18A and 18B

TABLE C-3 CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 30A AND 30B

GAGE SIZE NORMAL (W/O TABS)		P_{MAX} 28120 LBS	
ATTACHMENT FM-73		P_{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 HZ	
<u>CRACK GROWTH GAGE 30A</u>		<u>CRACK GROWTH GAGE 30B</u>	
<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>	<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>
EDM	0.100	EDM	0.102
PRECRACK	0.186	PRECRACK	0.314
1000	0.226	1000	0.375
2000	0.271	2000	0.432
3000	0.320	3000	0.482
4000	0.360	4000	0.548
5000	0.402	5000	0.621
6000	0.455	6000	0.689
7000	0.515	7000	0.774
8000	0.562	8000	0.865
9000	0.621	9000	0.958
10000	0.687	10000	1.061
11000	0.745	11000	1.163
12000	0.804	12000	1.304
13000	0.870		
14000	0.928		
15000	0.996		
16000	1.067		
18000	1.225		
19000	1.305		

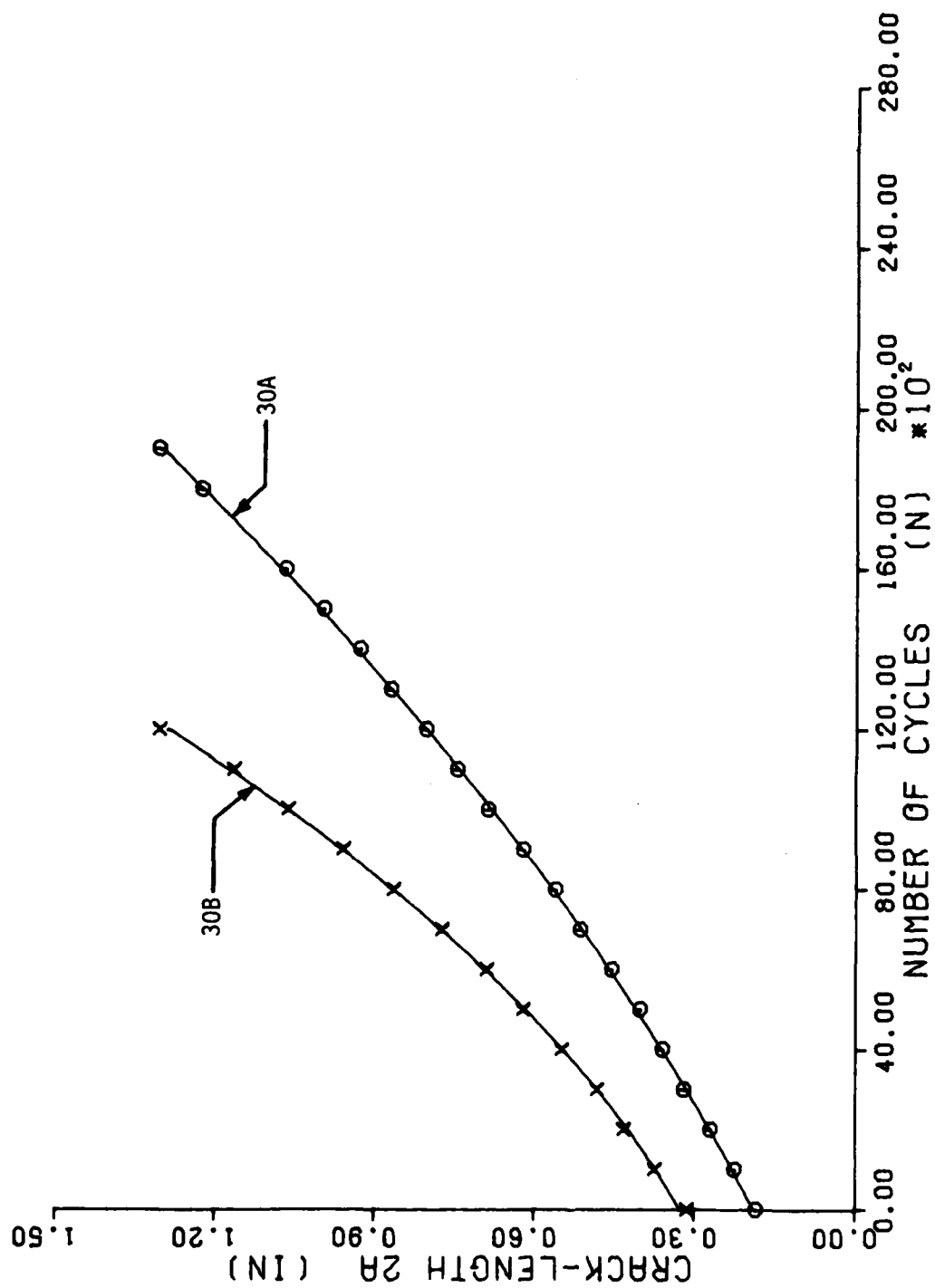


Figure C-3. Constant Amplitude Test of Crack Growth Gages 30A and 30B

TABLE C-4. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 24A AND 24B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 HZ	
CRACK GROWTH GAGE 24A		CRACK GROWTH GAGE 24B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.101	EDM	0.103
PRECRACK	0.1268	PRECRACK	0.233
1000	0.312	1000	0.274
2000	0.356	2000	0.317
3000	0.404	3000	0.367
4000	0.446	4000	0.421
5000	0.494	5000	0.477
6000	0.559	6000	0.532
7000	0.607	7000	0.597
8000	0.654	8000	0.663
9000	0.720	9000	0.729
10000	0.788	10000	0.793
11000	0.860	11000	0.865
12000	0.936	12000	0.939
13000	1.001	13000	1.028
14000	1.079	14000	1.131
15000	1.147	15000	1.236
16000	1.249		
17000	1.334		

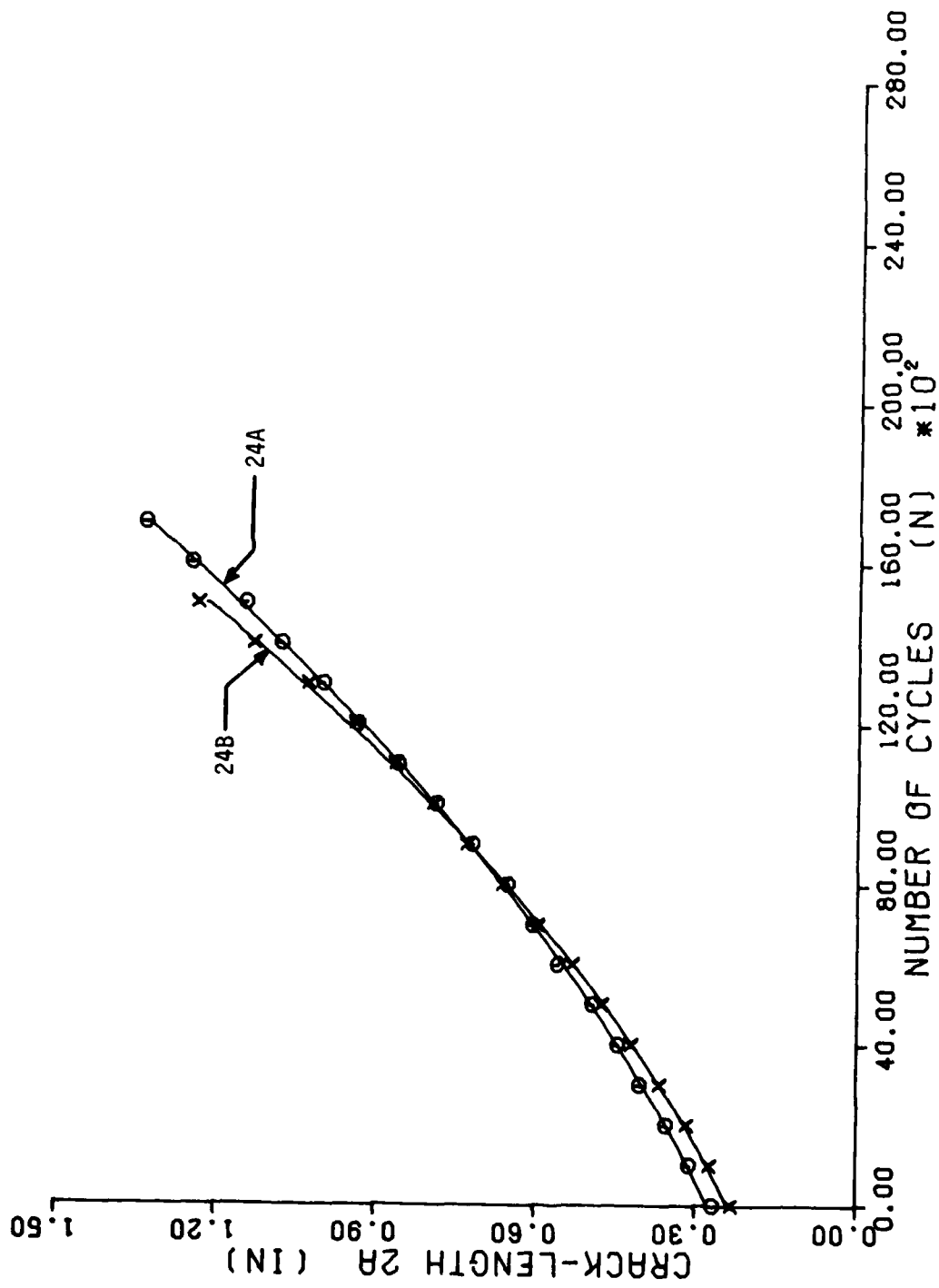


Figure C-4. Constant Amplitude Tests of Crack Growth Gages 24A and 24B

TABLE C-5. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES M2A AND M2B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 HZ	
CRACK GROWTH GAGE M2A		CRACK GROWTH GAGE M2B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.101	EDM	0.087
PRECRACK	0.143	PRECRACK	0.177
1000	0.169	1000	0.221
2000	0.205	2000	0.287
3000	0.204	3000	0.310
4000	0.272	4000	0.363
5000	0.320	5000	0.431
6000	0.371	6000	0.502
7000	0.430	7000	0.581
8000	0.498	8000	0.676
9000	0.564	9000	0.774
10000	0.640	10000	0.963
11000	0.736	11000	1.075
12000	0.821	12000	1.308
13000	0.913		
14000	1.027		
15000	1.140		
16000	1.308		

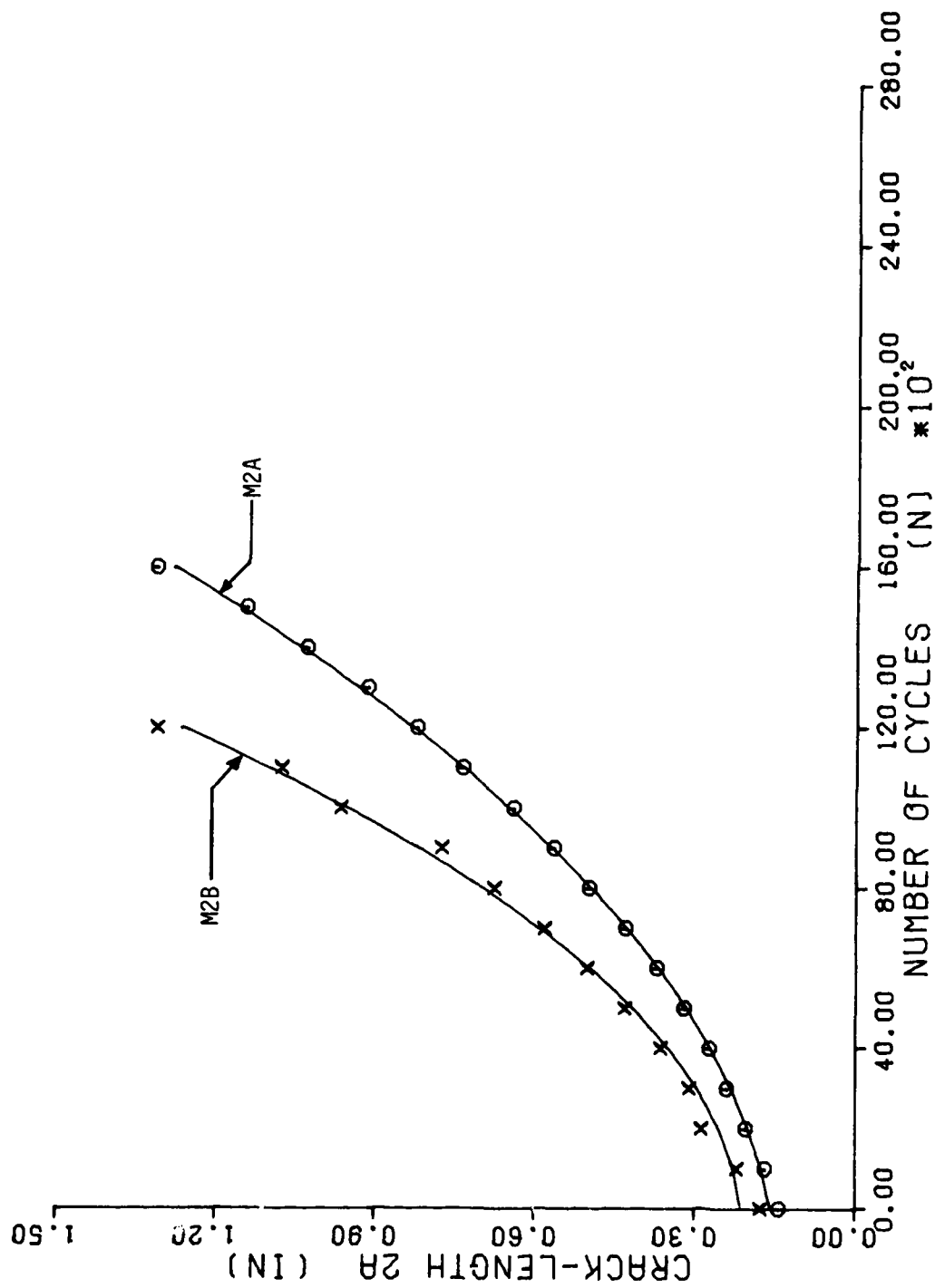


Figure C-5. Constant Amplitude Tests of Crack Growth Gages M2A and M2B

TABLE C-6. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES M4A AND M4B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE M4A		CRACK GROWTH GAGE M4B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.106	EDM	0.105
1000	0.111	1000	0.114
2000	0.141	2000	0.136
3000	0.163	3000	0.164
4000	0.191	4000	0.190
5000	0.223	5000	0.227
6000	0.260	6000	0.270
7000	0.292	7000	0.312
8000	0.339	8000	0.359
9000	0.403	9000	0.415
10000	0.446	10000	0.471
11000	0.500	11000	0.535
12000	0.558	12000	0.612
13000	0.653	13000	0.693
14000	0.736	14000	0.784
15000	0.834	15000	0.897
16000	0.890	16000	1.016
17000	0.967	17000	1.170
18000	1.168	18000	1.380

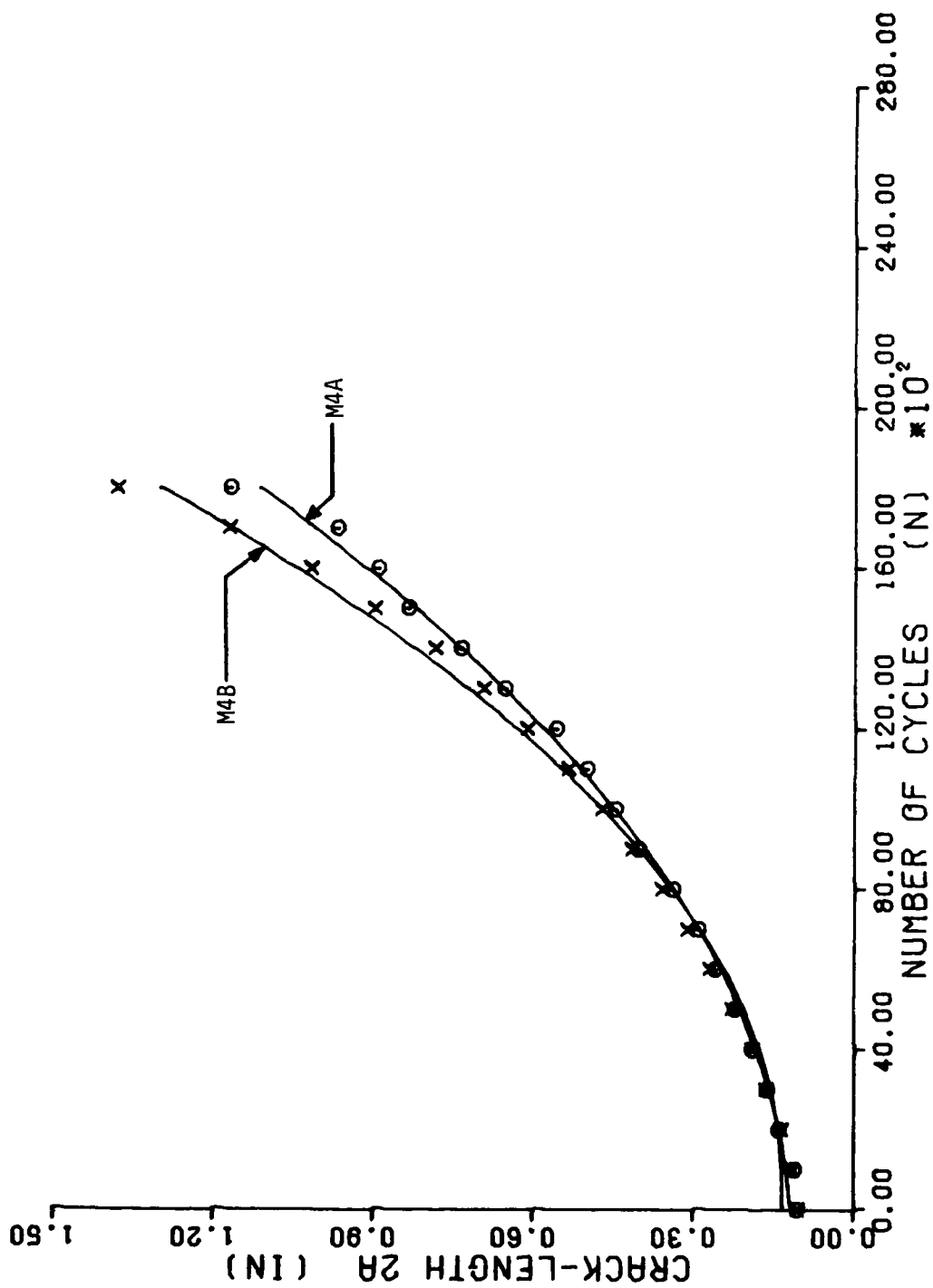


Figure C-6. Constant Amplitude Tests of Crack Growth Gages M4A and M4B

TABLE C-7. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 22A AND 22B

GAGE SIZE LONG (WITH TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73 & BOLTS		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE 22A		CRACK GROWTH GAGE 22B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.102	EDM	0.100
1000	0.110	1000	0.104
2000	0.122	2000	0.118
3000	0.142	3000	0.140
4000	0.167	4000	0.166
5000	0.194	5000	0.194
6000	0.223	6000	0.233
7000	0.254	7000	0.271
8000	0.290	8000	0.307
9000	0.336	9000	0.340
10000	0.378	10000	0.397
11000	0.427	11000	0.455
12000	0.479	12000	0.510
13000	0.538	13000	0.565
14000	0.593	14000	0.627
15000	0.658	15000	0.687
16000	0.726	16000	0.752
17000	0.802	17000	0.783
18000	0.878	18000	0.884
19000	0.926	19000	0.952
20000	1.008	20000	1.042
21000	1.091	21000	1.118
22000	1.182	22000	1.184
23000	1.269	23000	1.331

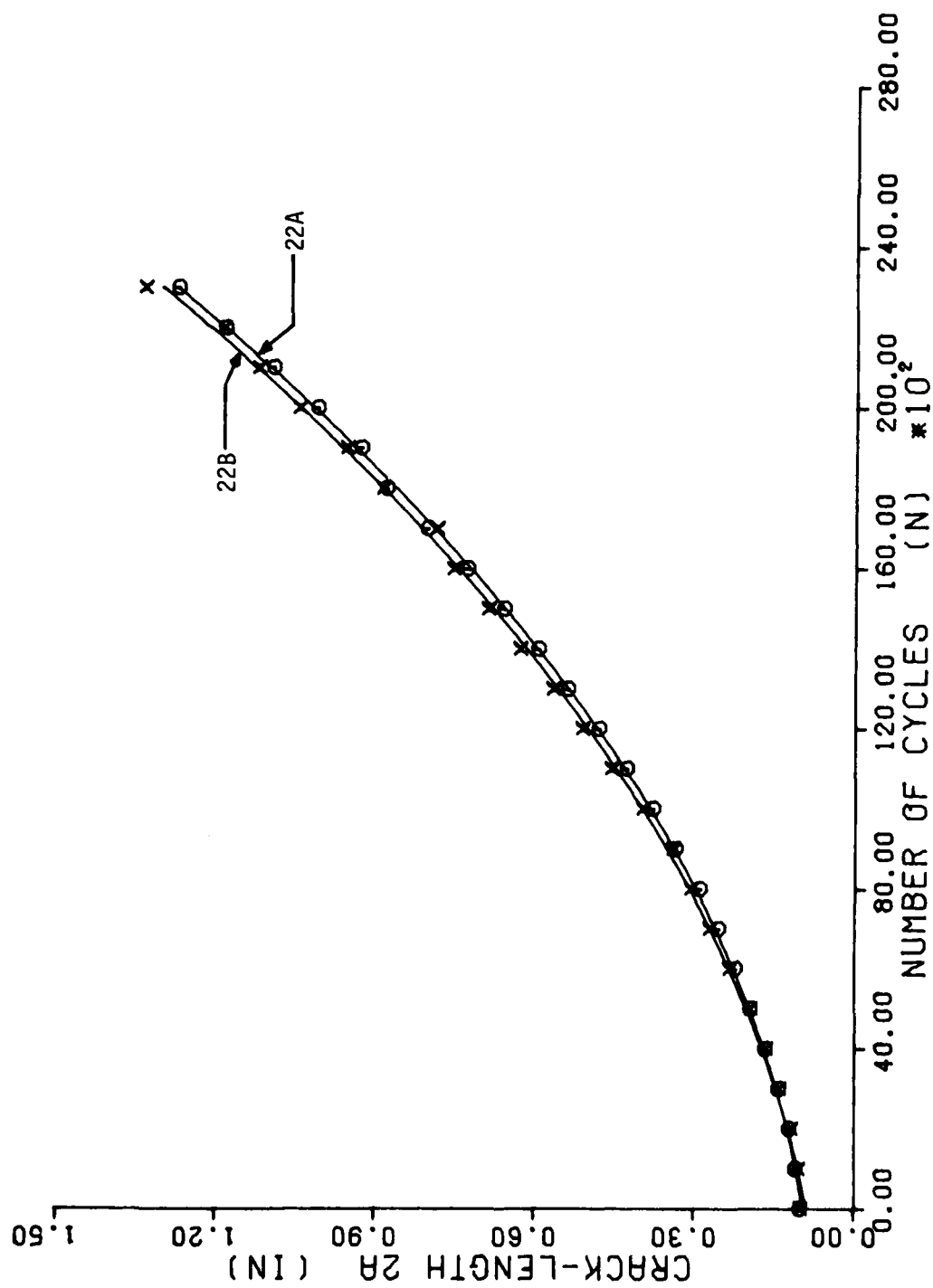


Figure C-7. Constant Amplitude Tests of Crack Growth Gages 22A and 22B

TABLE C-8. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 100A AND 100B

GAGE SIZE NORMAL (W/O TABS)		P_{MAX} 28120 LBS	
ATTACHMENT FM-73		P_{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE 100A		CRACK GROWTH GAGE 100B	
<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>	<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>
EDM	0.104	EDM	0.103
1000	0.216	1000	0.199
2000	0.258	2000	0.249
3000	0.297	3000	0.297
4000	0.341	4000	0.347
5000	0.387	5000	0.396
6000	0.446	6000	0.455
7000	0.501	7000	0.506
8000	0.556	8000	0.571
9000	0.613	9000	0.638
10000	0.681	1000	0.705
11000	0.753	11000	0.771
12000	0.827	12000	0.850
13000	0.919	13000	0.928
14000	1.011	14000	1.019
15000	1.098	15000	1.130
16000	1.195	16000	1.228
17000	1.296	17000	1.356

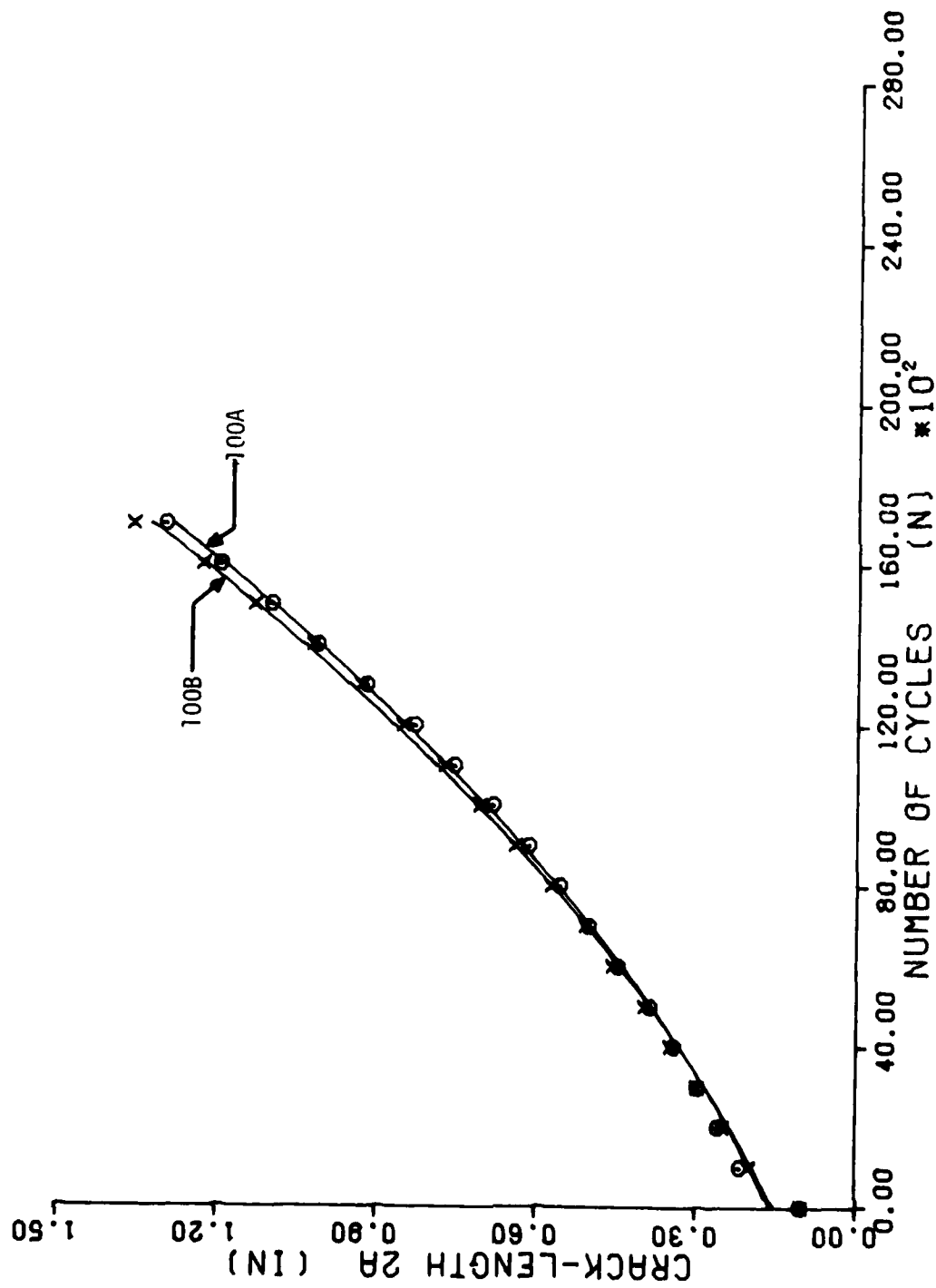


Figure C-8. Constant Amplitude Tests of Crack Growth Gages 100A and 100B

TABLE C-9. CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 48A AND 48B

GAGE SIZE LONG (WITH TABS)		P_{MAX} 28120 LBS	
ATTACHMENT FM-73		P_{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE 48A		CRACK GROWTH GAGE 48B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.102	EDM	0.100
1000	0.111	1000	0.107
2000	0.126	2000	1.129
3000	0.162	3000	0.158
4000	0.213	4000	0.189
5000	0.250	5000	0.224
6000	0.298	6000	0.262
7000	0.353	7000	0.302
8000	0.415	8000	0.348
9000	0.478	9000	0.397
10000	0.544	10000	0.447
11000	0.613	11000	0.503
12000	0.713	12000	0.555
13000	0.800	13000	0.610
14000	0.912	14000	0.678
15000	1.011	15000	0.747
16000	1.134	16000	0.808
17000		17000	0.873
18000		18000	0.938
19000		19000	0.994
20000		20000	1.073

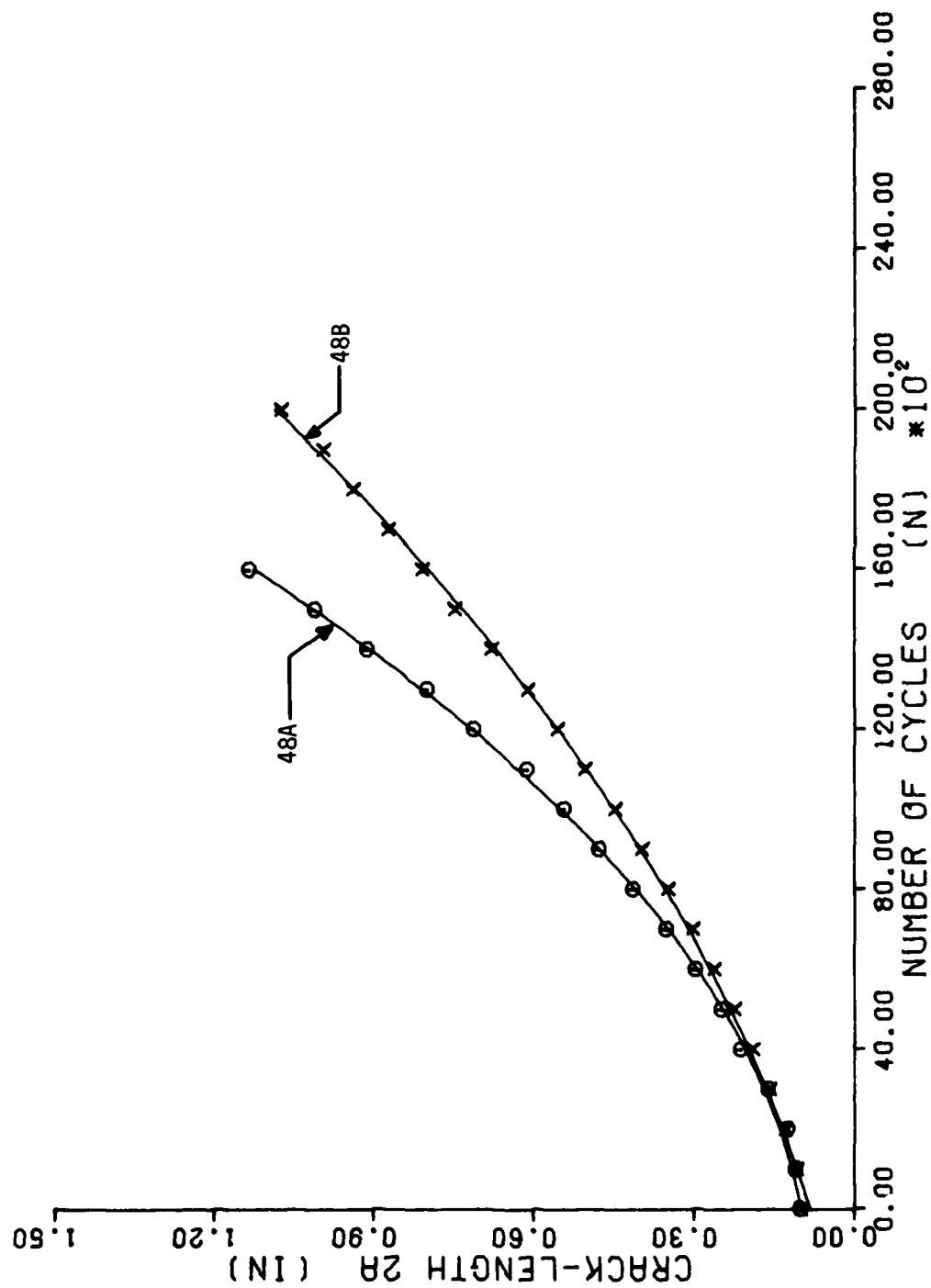


Figure C-9. Constant Amplitude Tests of Crack Growth Gages 48A and 48B

TABLE C-10 CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 32A AND 32B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT FM-73		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE 32A		CRACK GROWTH GAGE 32B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.103	EDM	0.104
PRECRACK	0.150	PRECRACK	0.159
300	0.191	300	0.200
1400	0.251	1400	0.253
2430	0.292	2430	0.302
3430	0.333	3430	0.305
4630	0.383	4630	0.402
5655	0.433	5655	0.449
6615	0.470	6615	0.500
7515	0.526	7515	0.556
8425	0.574	8425	0.604
9225	0.619	9225	0.656
9925	0.657	9925	0.699
10625	0.700	10625	0.748
11375	0.757	11375	0.799
12050	0.798	12050	0.852
12750	0.839	12750	0.901
13375	0.885	13375	0.950
13875	0.909	13875	0.998
14550	0.955	14550	1.051
15160	0.999	15160	1.099
15760	1.046	15760	1.150
16310	1.090	16310	1.210
17010	1.150	17010	1.276
17585	1.207	17585	1.358

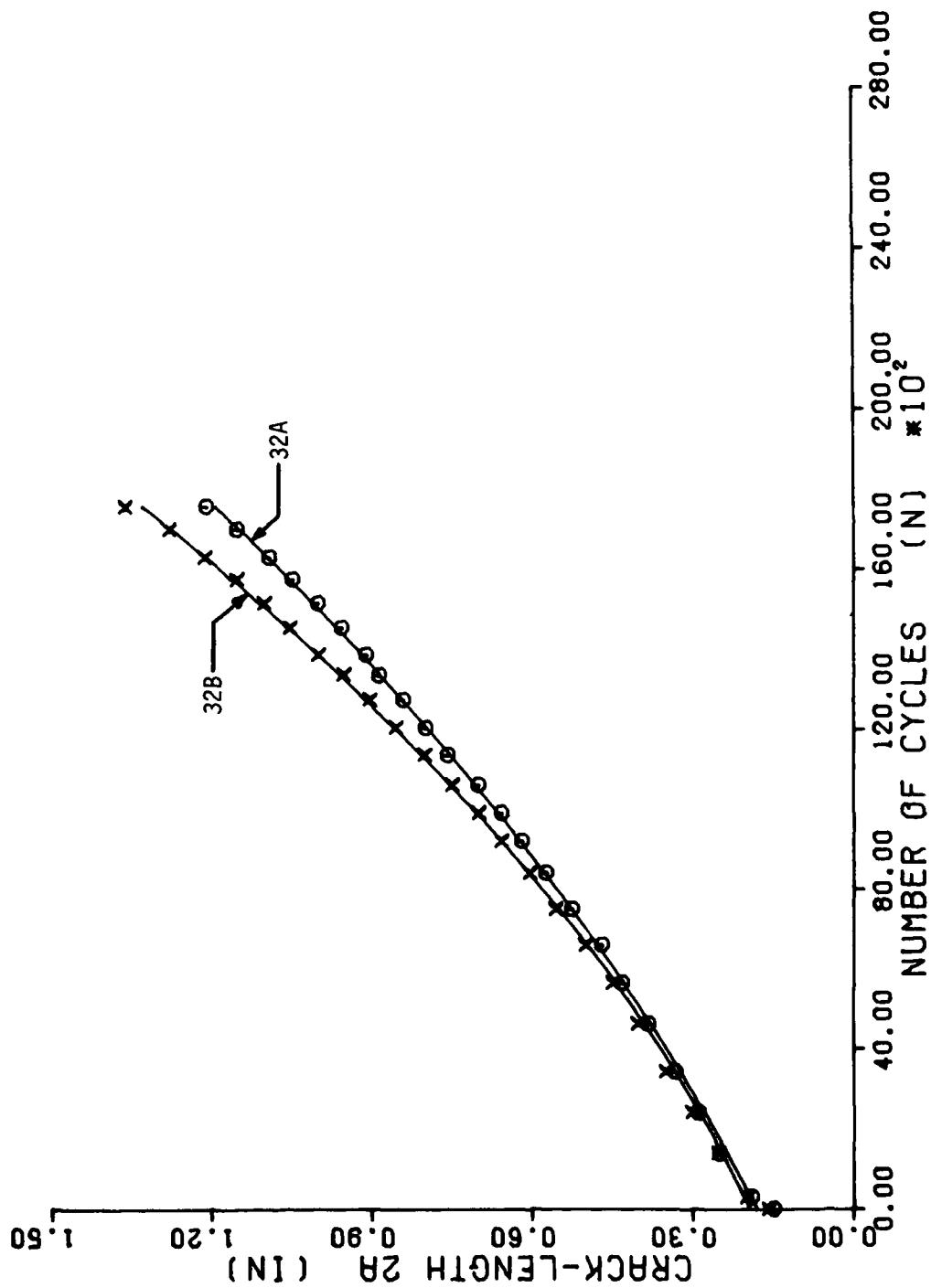


Figure C-10. Constant Amplitude Tests of Crack Growth Gages 32A and 32B

TABLE C-11 CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 118A AND 118B

GAGE SIZE NORMAL (W/O TABS)		P _{MAX} 28120 LBS	
ATTACHMENT EA 93091		P _{MIN} 0 LBS	
ENVIRONMENT LAB AIR		CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE 118A		CRACK GROWTH GAGE 118B	
CYCLES N	CRACK LENGTH 2a (INCHES)	CYCLES N	CRACK LENGTH 2a (INCHES)
EDM	0.105	EDM	0.105
2000	.133	2000	.121
3000	.155	3000	.140
4000	.182	4000	.159
5000	.211	5000	.190
6000	.246	6000	.208
7000	.282	7000	.239
8000	.321	8000	.262
9000	.366	9000	.302
10000	.423	10000	.329
11000	.445	11000	.364
12000	.507	12000	.403
13000	.565	13000	.443
14000	.614	14000	.479
15000	.679	15000	.513
16000	.751	16000	.553
17000	.822	17000	.603
18000	.902	18000	.649
19000	.983	19000	.696
20000	1.090	20000	.736
21000		21000	.789
22000		22000	.836
23000		23000	.885
24000		24000	.939
25000		25000	1.006
26000		26000	1.035
27000		27000	1.092

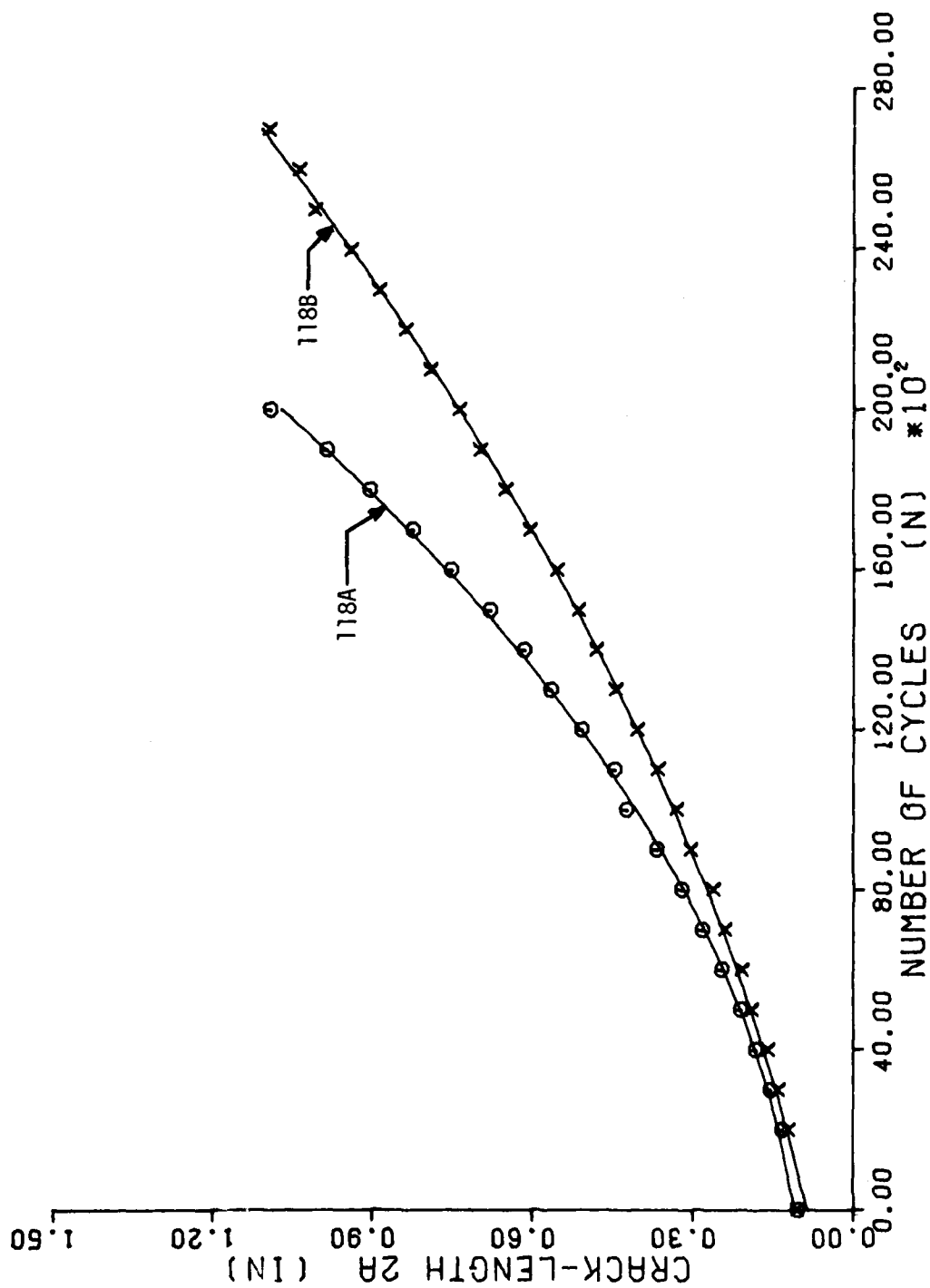


Figure C-11. Constant Amplitude Tests of Crack Growth Gages 118A and 118B

TABLE C-12 CRACK PROPAGATION DATA FOR CRACK GROWTH GAGES 110A AND 110B

GAGE SIZE	LONG (WITH TABS)	P_{MAX} 28120 LBS	
ATTACHMENT	EA 93909J	P_{MIN} 0 LBS	
ENVIRONMENT	LAB AIR	CYCLIC RATE 0.5 Hz	
CRACK GROWTH GAGE 110A		CRACK GROWTH GAGE 110B	
<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>	<u>CYCLES</u> <u>N</u>	<u>CRACK LENGTH</u> <u>2a (INCHES)</u>
EDM	0.103	EDM	0.103
2000	.125	2000	.119
3000	.144	3000	.157
4000	.172	4000	.185
5000	.199	5000	.223
6000	.229	6000	.263
7000	.271	7000	.312
8000	.312	8000	.366
9000	.352	9000	.425
10000	.398	10000	.491
11000	.434	11000	.567
12000	.483	12000	.656
13000	.546	13000	.745
14000	.600	14000	.835
15000	.659	15000	.942
16000	.718	16000	1.050
17000	.785		
18000	.855		
19000	.908		

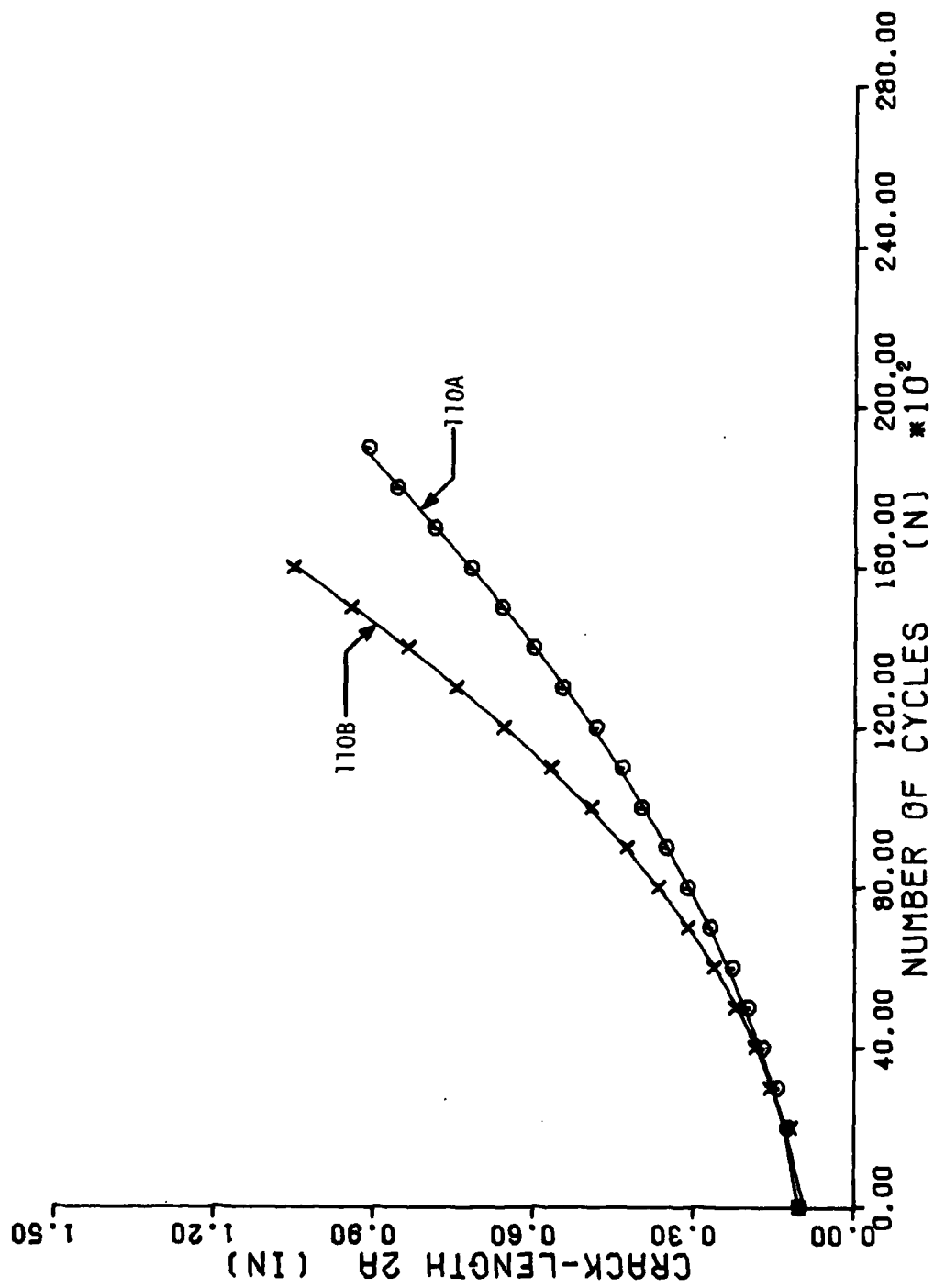


Figure C-12. Constant Amplitude Tests of Crack Growth Gages 110A and 110B

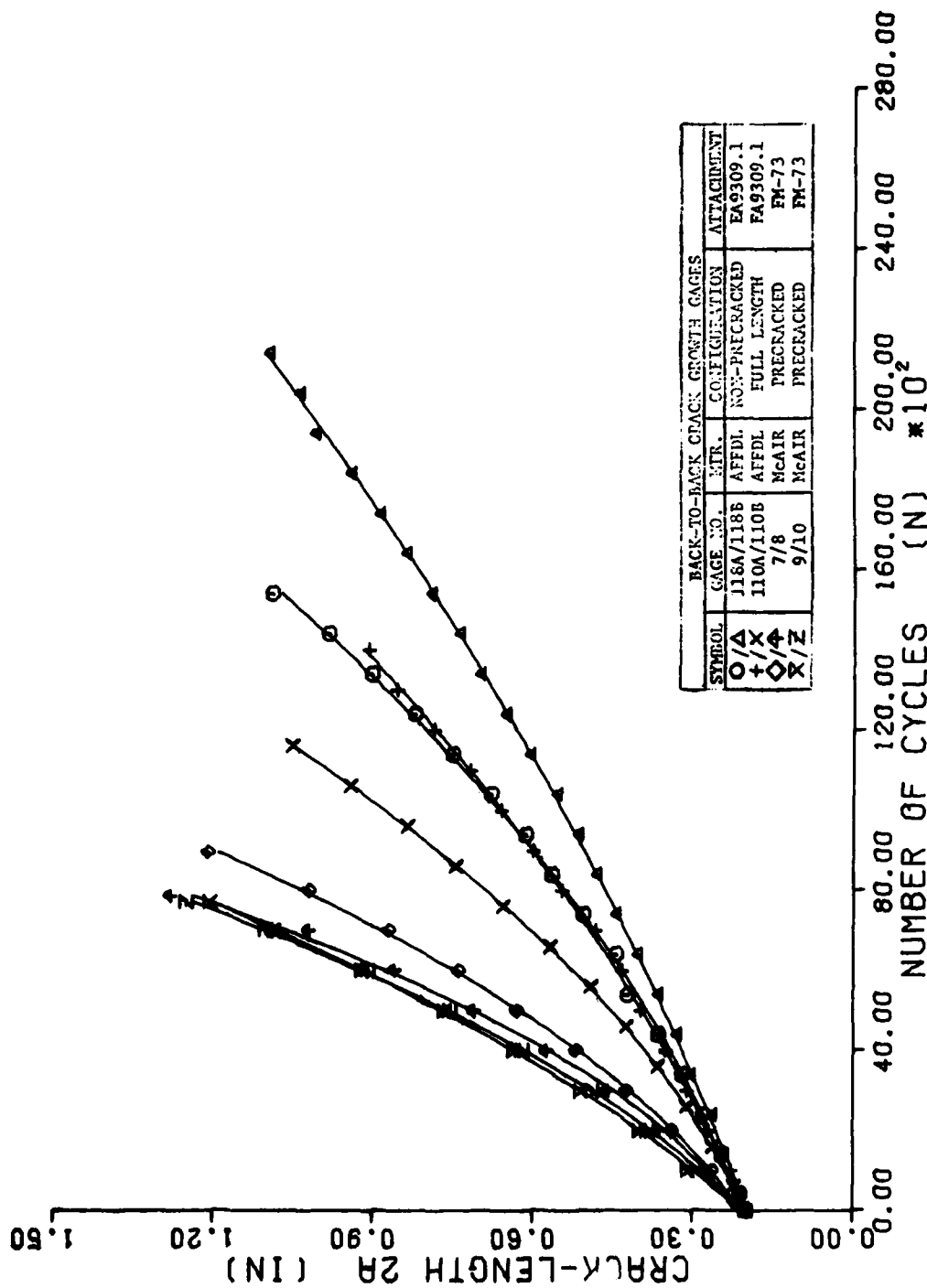


Figure C-13. Constant Amplitude Tests - AFFDL Tests of AFFDL Gages (Bonded with EA9309.1 Adhesive) Compared with McAir Tests of McAir Gages (Bonded with FM-73 Adhesive)

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